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NASA/SP—1998-7037/SUPPL387
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Typical Report Citation and Abstract

① 19970001126 NASA Langley Research Center, Hampton, VA USA
② Water Tunnel Flow Visualization Study Through Poststall of 12 Novel Planform Shapes
③ Gatlin, Gregory M., NASA Langley Research Center, USA Neuhart, Dan H., Lockheed Engineering and Sciences Co., USA;
④ Mar. 1996; 130p; In English
⑤ Contract(s)/Grant(s): RTOP 505-68-70-04
⑥ Report No(s): NASA-TM-4663; NAS 1.15:4663; L-17418; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche
⑦ To determine the flow field characteristics of 12 planform geometries, a flow visualization investigation was conducted in the Langley 16- by 24-Inch Water Tunnel. Concepts studied included flat plate representations of diamond wings, twin bodies, double wings, cutout wing configurations, and serrated forebodies. The off-surface flow patterns were identified by injecting colored dyes from the model surface into the free-stream flow. These dyes generally were injected so that the localized vortical flow patterns were visualized. Photographs were obtained for angles of attack ranging from 10° to 50°, and all investigations were conducted at a test section speed of 0.25 ft per sec. Results from the investigation indicate that the formation of strong vortices on highly swept forebodies can improve poststall lift characteristics; however, the asymmetric bursting of these vortices could produce substantial control problems. A wing cutout was found to significantly alter the position of the forebody vortex on the wing by shifting the vortex inboard. Serrated forebodies were found to effectively generate multiple vortices over the configuration. Vortices from 65° swept forebody serrations tended to roll together, while vortices from 40° swept serrations were more effective in generating additional lift caused by their more independent nature.
⑧ Author
⑨ *Water Tunnel Tests; Flow Visualization; Flow Distribution; Free Flow; Planforms; Wing Profiles; Aerodynamic Configurations*

Key

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AERONAUTICAL ENGINEERING

A Continuing Bibliography (Suppl. 387)

NOVEMBER 13, 1998

01
AERONAUTICS

19980227102 NASA Langley Research Center, Hampton,VA USA
Aeronautical Engineering: A Continuing Bibliography, Supplment 385
Oct. 16, 1998; 42p; In English
Report No.(s): NASA/SP-1998-7037/SUPPL385; NAS 1.21:7037/SUPPL385; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This supplemental issue of Aeronautical Engineering, A Continuing Bibliography with Indexes (NASA/SP-1998-7037) lists reports, articles, and other documents recently announced in the NASA STI Database. The coverage includes documents on the engineering and theoretical aspects of design, construction, evaluation, testing, operation, and performance of aircraft (including aircraft engines) and associated components, equipment, and systems. It also includes research and development in aerodynamics, aeronautics, and ground support equipment for aeronautical vehicles. Each entry in the publication consists of a standard bibliographic citation accompanied, in most cases, by an abstract.
CASI
Bibliographies; Aerodynamics; Aeronautical Engineering; Indexes (Documentation); Aircraft Design

19980227425 Carnegie-Mellon Inst. of Research, Pittsburgh, PA USA
Automated Inspection of Aircraft *Final Report*
Alberts, C. J., Carnegie-Mellon Inst. of Research, USA; Carroll, C. W., Carnegie-Mellon Inst. of Research, USA; Kaufman, W. M., Carnegie-Mellon Inst. of Research, USA; Perlee, C. J., Carnegie-Mellon Inst. of Research, USA; Siegel, M. W., Carnegie-Mellon Inst. of Research, USA; Apr. 1998; 85p; In English
Contract(s)/Grant(s): FAA94-G-018
Report No.(s): AD-A350525; AAR-430; DOT/FAA/AR-97/69; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This report summarizes the development of a robotic system designed to assist aircraft inspectors by remotely deploying nondestructive inspection (NDI) sensors and acquiring, processing, and storing inspection data. Carnegie Mellon University studied the task of aircraft inspection, compiled the functional requirements for an automated system to inspect skin fastener rows, and developed a conceptual design of an inspection robot. A prototype of the robotic inspection system (the Automated Nondestructive Inspector or (ANDI) was developed. The first phase of system development resulted in a laboratory system that demonstrated the abilities to adhere to the surface of an aircraft panel and deploy a standard eddy-current sensor. The second phase of development included enhancing the mechanics, adding video cameras to the robot for navigation, and adding an on-board computer for low-level task sequencing. The second-phase system was subsequently demonstrated at the FAA's Aging Aircraft NDI Validation Center (AANC). During the final phase of development, emphasis was placed on the enhancement of the robot's navigational system through automated recognition of image features captured by the navigation cameras. A significant development effort remains to be accomplished before this robotic inspection technology is suitable for operational deployment. Outstanding development issues include: (1) reducing the weight of the robot so that it is more comfortable to lift and position on the aircraft; (2) improving the mechanical reliability and speed of the system; (3) minimizing the scratching of the skin surface by the suction cups and eddy-current sensors; (4) reduction or elimination of the umbilical cable; and (5) automation of the manually controlled operations. to commercialize the technology, a new mechanical system would need to be designed and built incorporating the lessons of this work.
DTIC
Inspection; Aircraft Structures; Aircraft Maintenance; Robot Control; Robotics; Nondestructive Tests

02
AERODYNAMICS

Includes aerodynamics of bodies, combinations, wings, rotors, and control surfaces; and internal flow in ducts and turbomachinery.

19980221789 European Organization for the Safety of Air Navigation, Bretigny-sur-Orge, France
User Manual for the Base of Aircraft Data (BADA)
Bos, A., European Organization for the Safety of Air Navigation, France; Mar. 1998; 100p; In English
Report No.(s): PB98-164312; EEC/NOTE-6/98-Rev-3.0; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

The BASE of Aircraft Data (BADA) provides a set of ASCII files containing performance and operating procedure coefficients for 151 different aircraft types. The coefficients include those used to calculate thrust, drag and fuel flow and those used to specify nominal cruise, climb and decent speed. User Manual for Revision 3.0 of BADA provides definitions of each of the coefficients and then explains the file formats. Instructions for remotely accessing the files via Internet are also given.
NTIS
Aerodynamic Drag; User Manuals (Computer Programs); Climbing Flight

19980223077 NASA Ames Research Center, Moffett Field, CA USA
Lift, Drag, Static Stability and Control Characteristics and Control Surface Panel Loads From Wind Tunnel Tests at Supersonic Speeds of Models of Two Versions of the B-70 Airplane
Daugherty, James C., NASA Ames Research Center, USA; Green, Kendal H., NASA Ames Research Center, USA; Nelson, Richard D., NASA Ames Research Center, USA; Oct. 1960; 148p; In English
Report No.(s): NASA-TM-SX-396; X68-84368; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

Models of two versions of the North American B-70 airplane were tested at Mach numbers of 2.5 and 3.0 at a Reynolds number of 5.0 million and at a Mach number of 3.5 at a Reynolds number of 4.5 million. Lift, drag, static-stability, and control characteristics were determined for both complete models, for various components, and for various component modifications. Flow visualization studies were made with both natural and fixed boundary-layer transition. The data for the best configuration tested were extrapolated to give an all-turbulent lift-drag ratio for a Reynolds number of 110 million at a Mach number of 3.0.
Author
Wind Tunnel Tests; B-70 Aircraft; Boundary Layer Transition; Flow Visualization; Mach Number; Reynolds Number; Supersonic Speed; Lift; Drag; Static Stability; Controllability; Dynamic Characteristics

19980223577 NASA Langley Research Center, Hampton, VA USA
Analytic Study of Induced Pressure on Long Bodies of Revolution with Varying Nose Bluntness at Hypersonic Speeds
VanHise, Vernon, NASA Langley Research Center, USA; 1961; 20p; In English
Report No.(s): NASA-TR-R-78; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Pressure distributions and shock shapes for a series of cylindrical afterbodies having nose fineness ratios from 0.4 to 4 have been calculated by using the method of characteristics for a perfect gas. The fluid mediums investigated were air and helium and the Mach number range was from 5 to 40. Flow parameters obtained from blast-wave analogy gave good correlations of blunt-nose induced pressures and shock shapes. Experimental results are found to be in good agreement with the characteristic calculations. The concept of hypersonic similitude enables good correlation of the results with respect to body shape, Mach number, and ratio of specific heats.
Author
Hypersonic Speed; Pressure Distribution; Cylindrical Bodies; Bodies of Revolution; Afterbodies; Hypersonics; Method of Characteristics; Blunt Bodies; Shock Waves

19980223578 NASA Ames Research Center, Moffett Field, CA USA
Effects of Outboard Thickened and Blunted Leading Edges on the Wave Drag of a 45 Degree Swept-Wing and Body Combination
Holdaway, George H., NASA Ames Research Center, USA; Lazzeroni, Frank A., NASA Ames Research Center, USA; Hatfield, Elaine W., NASA Ames Research Center, USA; Aug. 1959; 34p; In English
Report No.(s): NASA-TM-X-27; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation to evaluate the effects of thickened and blunted leading-edge modifications on the wave drag of a swept wing has been made at Mach numbers from 0.65 to 2.20 and at a Reynolds number of 2,580,000 based on the mean aerodynamic chord of the basic wing. Two leading-edge designs were investigated and they are referred to as the thickened and the blunted modifications although both sections had equally large leading-edge radii. The thickened leading edge was formed by increasing the thick-

ness over the forward 40 percent of the basic wing section. The blunted modification was formed by reducing the wing chords about 1 percent and by increasing the section thickness slightly over the forward 6 percent of the basic section in a manner to keep the wing sweep and volume essentially equal to the respective values for the basic wing. The basic wing had an aspect ratio of 3, a leading-edge sweep of 45 deg., a taper ratio of 0.4, and NACA 64A006 sections perpendicular to a line swept back 39.45 deg., the quarter-chord line of these sections. Test results indicated that the thickened modification resulted in an increase in zero-lift drag coefficient of from 0.0040 to 0.0060 over values for the basic model at Mach numbers at which the wing leading edge was sonic or supersonic. Although drag coefficients of both the basic and thickened models were reduced at all test Mach numbers by body indentations designed for the range of Mach numbers from 1.00 to 2.00, the greater drag of the thickened model relative to that of the basic model was not reduced. The blunted model, however, had less than one quarter of the drag penalty of the thickened model relative to the basic model at supersonic leading-edge conditions (M greater or equal to root-2).
Author
Swept Wings; Blunt Bodies; Aerodynamic Coefficients; Airfoil Profiles; Leading Edges; Thickness; Wave Drag; Wind Tunnel Tests; Wind Tunnel Models

19980223579 NASA Langley Research Center, Hampton, VA USA
Large Angle Motion Tests, Including Spins, of a Free-Flying Radio-Controlled 0.13-Scale Model of a Twin-Jet Swept-Wing Fighter Airplane
Burk, Sanger M., Jr., NASA Langley Research Center, USA; Libbey, Charles E., NASA Langley Research Center, USA; 1961; 42p; In English
Report No.(s): NASA-TM-SX-445; L-1192; N-AM-50; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted with a free-flying radio-controlled 0.13 scale model of a twin-jet swept-wing fighter airplane to determine the tendency of this design to enter spins and to evaluate the nature of the spin obtained from post-stall motions. The test results indicate that it may be difficult to obtain a developed spin on the airplane, particularly the flat-type spin. Two types of erect developed spins will be possible: one will be flat and fast rotating from which recovery may not be obtained and the other will be steeper and oscillatory from which recoveries will be satisfactory. Controls will be effective for satisfactory termination of the post-stall gyrations obtained. The recommended recovery technique from both post-stall gyrations and developed spins will be movement of the rudder to oppose the yawing rotation and simultaneous movement of the ailerons to with the rotation (stick right when turning to the right). When recovery is imminent, the stick should be moved longitudinally to neutral. It is recommended that the spin not be allowed to develop fully on this airplane. The developed-spin results obtained in the investigation were in good agreement with spin-tunnel results.
Author
Fighter Aircraft; Aircraft Spin; Spin Dynamics; Aerodynamic Stalling; Wind Tunnel Tests; Wind Tunnel Models; Scale Models; Swept Wings; Control Stability; Maneuvers; Aircraft Stability

19980223580 NASA Langley Research Center, Hampton, VA USA
Free-Spinning-Tunnel Investigation of a 1/30 Scale Model of a Twin-Jet-Swept-Wing Fighter Airplane
Bowman, James S., Jr., NASA Langley Research Center, USA; Healy, Frederick M., NASA Langley Research Center, USA; 1960; 22p; In English
Report No.(s): NASA-TM-SX-446; L-1191; N5154; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made in the Langley 20-foot free-spinning tunnel to determine the erect and inverted spin and recovery characteristics of a 1/30-scale dynamic model of a twin-jet swept-wing fighter airplane. The model results indicate that the optimum erect spin recovery technique determined (simultaneous rudder reversal to full against the spin and aileron deflection to full with the spin) will provide satisfactory recovery from steep-type spins obtained on the airplane. It is considered that the air-plane will not readily enter flat-type spins, also indicated as possible by the model tests, but developed-spin conditions should be avoided in as much as the optimum recovery procedure may not provide satisfactory recovery if the airplane encounters a flat-type developed spin. Satisfactory recovery from inverted spins will be obtained on the airplane by neutralization of all controls. A 30-foot- diameter (laid-out-flat) stable tail parachute having a drag coefficient of 0.67 and a towline length of 27.5 feet will be satisfactory for emergency spin recovery.
Author
Aircraft Spin; Control Stability; Spin Dynamics; Wind Tunnel Tests; Wind Tunnel Models; Aerodynamic Stalling; Fighter Aircraft; Scale Models; Swept Wings; Aircraft Stability

19980223582 **NASA Langley Research Center, Hampton, VA USA**

Summary of Results Obtained in Full-Scale Tunnel Investigation of the Ryan Flex-Wing Airplane

Johnson, Joseph L., Jr., NASA Langley Research Center, USA; Hassell, James L., Jr., NASA Langley Research Center, USA; Aug. 14, 1962; 40p; In English

Report No.(s): NASA-TM-SX-727; L-3093; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The performance and static stability and control characteristics of the Ryan Flex-Wing airplane were determined in an investigation conducted in the Langley full-scale tunnel through an angle-of-attack range of the keel from about 14 to 44 deg. for power-on and -off conditions. Comparisons of the wind-tunnel data with flight-test data obtained with the same airplane by the Ryan Aeronautical Company were made in a number of cases.

Author

Wind Tunnel Tests; Static Stability; Aircraft Performance; Wings; Longitudinal Stability; Aircraft Stability; Flight Characteristics

19980223585 **NASA Langley Research Center, Hampton, VA USA**

An Investigation of the Influence of Body Size and Indentation Asymmetry of the Effectiveness of Body Indentation in Combination with a Cambered Wing

Patterson, James C., Jr., NASA Langley Research Center, USA; Loving, Donald L., NASA Langley Research Center, USA; Feb. 1961; 38p; In English

Report No.(s): NASA-TM-X-427; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made of a 450 sweptback cambered wing in combination with an unindented body and a body symmetrically indented with respect to its axes designed for a Mach number of 1.2. The ratio of body frontal area to wing planform area was 0.08 for these wing-body combinations. In order to determine the influence of body size on the effectiveness of indentation, the test data have been compared with previously obtained data for similar configurations having a ratio of body frontal area to wing planform area of 0.04. Also, in order to investigate the relative effectiveness of indentation asymmetry, a specially indented body designed to account for the wing camber and also designed for a Mach number of 1.2 has been included in these tests. The investigation was conducted in the Langley 8-Foot Tunnels Branch at Mach numbers from 0.80 to 1.43 and a Reynolds number of approximately 1.85 x 10(exp 6), based on a mean aerodynamic chord length of 5.955 inches. The data indicate that the configurations with larger ratio of body frontal area to wing planform area had smaller reductions in zero-lift wave drag associated with body indentation than the configurations with smaller ratio of body frontal area to wing planform area. The 0.08-area-ratio configurations also had correspondingly smaller increases in the values of maximum lift-drag ratio than the 0.04-area-ratio configurations. The consideration of wing camber in the body indentation design resulted in a 35.5-percent reduction in zero-lift wave drag, compared with a 21.5-percent reduction associated with the symmetrical indentation, but had a negligible effect on the values of maximum lift-drag ratio.

Author

Body-Wing Configurations; Lift Drag Ratio; Indentation; Cambered Wings; Airfoil Profiles; Sweptback Wings

19980223586 **NASA Dryden Flight Research Center, Edwards, CA USA**

Preliminary Full-Scale Power-Off Drag of the X-15 Airplane for Mach Numbers from 0.7 to 3.1

Saltzman, Edwin J., NASA Dryden Flight Research Center, USA; Dec. 1960; 26p; In English

Report No.(s): NASA-TM-X-430; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Drag characteristics have been obtained for the X-15 airplane during unpowered flight. These data represent a Mach number range from about 0.7 to 3.1 and a Reynolds number range from 13.9 x 10(exp 6) to 28 x 10(exp 8), based on the mean aerodynamic chord. The full-scale data are compared with estimates compiled from several wind-tunnel facilities. The agreement between wind-tunnel and full-scale supersonic drag, uncorrected for Reynolds number effects, is reasonably close except at low supersonic Mach numbers where the flight values are significantly higher.

Author

X-15 Aircraft; Aerodynamic Drag; Mach Number; Airfoil Profiles; Supersonic Drag; Wind Tunnel Tests

19980223590 **NASA Langley Research Center, Hampton, VA USA**

Several Methods for Aerodynamic Reduction of Static-Pressure Sensing Errors for Aircraft at Subsonic, Near-Sonic, and Low Supersonic Speeds

Ritchie, Virgil S., NASA Langley Research Center, USA; 1959; 28p; In English

Report No.(s): NASA-TR-R-18; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were conducted in transonic wind tunnels to investigate and verify experimentally methods for aerodynamically reducing errors due to sensor position, bow-wave passage, and angle of attack. The results indicated that aerodynamic devices of simple

design may be employed to reduce errors in sensing static pressures to less than 0.5 percent at Mach numbers from about 0.40 to 1.15.
Author
Pressure Reduction; Aerodynamics; Static Pressure; Angle of Attack; Errors; Detection; Bow Waves

19980223594 **NASA Langley Research Center, Hampton, VA USA**
Ordinates and Theoretical Pressure-Distribution Data for NACA 6- and 6A-Series Airfoil Sections with Thicknesses from 2 to 21 and From 2 to 15 Percent Chord, Respectively
Patterson, Elizabeth W., NASA Langley Research Center, USA; **Braslow, Albert L.**, NASA Langley Research Center, USA; 1961; 88p; In English
Report No.(s): NASA-TR-R-84; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Information is presented with which ordinates can be easily obtained for any thickness from 2 to 21 percent chord for NACA 63-, 64-, and 65-series airfoil sections and from 2 to 15 percent chord for NACA 63A-, 64-A, series airfoil sections. In addition, data required for estimation of the theoretical pressure distributions of any of these airfoils are included.
Author
Airfoils; Pressure Distribution; Airfoil Profiles; Aircraft Design

19980223602 **NASA Ames Research Center, Moffett Field, CA USA**
Lift, Drag, and Pitching Moments of an Arrow Wing Having 80 Degree of Sweepback at Mach Numbers from 2.48 to 3.51 and Reynolds Numbers up to 11.0 Million
Hopkins, Edward J., NASA Ames Research Center, USA; **Jillie, Don W.**, NASA Ames Research Center, USA; **Levin, Alan D.**, NASA Ames Research Center, USA; Aug. 1959; 46p; In English
Report No.(s): NASA-TM-X-22; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Measurements were made of the lift, drag, and pitching moments on an arrow wing (taper ratio of zero) having an aspect ratio of 1.4 and a leading-edge sweepback of 80 (degrees). The wing was designed to have a subsonic leading-edge and a Clark-Y airfoil with a thickness ratio of 12 percent of the chord perpendicular to the wing leading edge. The wing was tested both with and without the wing tips bent upward in an attempt to alleviate possible flow separation in the vicinity of the wing tips. Small jets of air were used to fix transition near the wing leading edge. Force results are presented for Mach numbers of 2.48, 2.75, 3.04, 3.28, and 3.51 at Reynolds numbers of 3.5 and 9.0 million and for a Mach number of 3.04 at a Reynolds number of 11.0 million. The measured aerodynamic characteristics are compared with those estimated by linear theory. The maximum lift-drag ratio measured was much less than that predicted. This difference is attributed to lack of full leading-edge thrust and to the experimental lift-curve slope being about 20 percent below the theoretical value.
Author
Lift; Aerodynamic Drag; Pitching Moments; Separated Flow; Wind Tunnel Tests; Sweptback Wings; Boundary Layer Separation; Arrow Wings

19980223603 **NASA Langley Research Center, Hampton, VA USA**
Effect of Multiple-Jets Exits on the Base Pressure of a Simple Wing-Body Combination at Mach Numbers of 0.6 to 1.27
Cubbage, James M., Jr., NASA Langley Research Center, USA; Aug. 1959; 38p; In English
Report No.(s): NASA-TM-X-25; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted at Mach numbers of 0.6 to 1.27 to determine the effect of multiple-jet exits on the base pressure of a simple wing-body combination. The design Mach number of the nozzles ranged from 1 to 3 at jet exit diameters equal to 36.4 to 75 percent of the model thickness. Jet total-pressure to free-stream static-pressure ratios ranged from 1 (no flow) to 34.2. The results show that the variation of base pressure coefficient with jet pressure ratio for the model tested was similar to that obtained for single nozzles in bodies of revolution in other investigations. As in the case for single jets the base pressure coefficient for the present model became less negative as the jet exit diameter increased. For a constant throat diameter and an assumed schedule of jet pressure ratio over the speed range of these tests, nozzle Mach number had only a small effect on base pressure coefficient.
Author
Body-Wing Configurations; Base Pressure; Transonic Speed; Jet Exhaust; Free Flow; Exhaust Nozzles

19980223605 **NASA Langley Research Center, Hampton, VA USA**
A Supersonic Area Rule and an Application to the Design of a Wing-Body Combination with High Lift-Drag Ratios
Whitcomb, Richard T., NASA Langley Research Center, USA; **Sevier, John R., Jr.**, NASA Langley Research Center, USA; 1960; 16p; In English

Report No.(s): NASA-TR-R-72; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A concept for interrelating the wave drags of wing-body combinations at supersonic speeds with axial developments of cross-sectional area is presented. A swept-wing-indented-body combination designed on the basis of this concept to have significantly improved maximum lift-drag ratios over a range of transonic and moderate supersonic speeds is described. Experimental results have been obtained for this configuration at Mach numbers from 0.80 to 2.01. Maximum lift-drag ratios of approximately 14 and 9 were measured at Mach numbers of 1.15 and 1.41, respectively.

Author

Body-Wing Configurations; Supersonic Speed; Wave Drag; Transonic Speed; Lift Drag Ratio

19980223609 NASA Langley Research Center, Hampton, VA USA

Calculated Effects of Body Shape on the Bow-Shock Overpressures in the Far Field of Bodies in Supersonic Flow

Lansing, Donald L., NASA Langley Research Center, USA; 1960; 16p; In English

Report No.(s): NASA-TR-R-76; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A theory for the supersonic flow about bodies in uniform flight in a homogeneous medium is reviewed and an integral which expresses the effect of body shape upon the flow parameters in the far field is reduced to a form which may be readily evaluated for arbitrary body shapes. This expression is then used to investigate the effect of nose angle, fineness ratio, and location of maximum body cross section upon the far-field pressure jump across the bow-shock of slender bodies. Curves are presented showing the variation of the shock strength with each of these parameters. It is found that, for a wide variety of shapes having equal fineness ratios, the integral has nearly a constant value.

Author (revised)

Slender Bodies; Bow Waves; Sonic Booms; Supersonic Flight; Jet Aircraft Noise; Aircraft Structures; Aerodynamic Noise; Aircraft Design; Overpressure

19980223614 NASA Langley Research Center, Hampton, VA USA

Longitudinal Aerodynamic Characteristics of a Wing-Body-Tail Model Having a Highly Tapered, Cambered 45 degree Swept Wing of Aspect Ratio 4 at Transonic Speeds

West, F. E., Jr., NASA Langley Research Center, USA; Nov. 1959; 32p; In English

Report No.(s): NASA-TM-X-130; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The longitudinal aerodynamic characteristics of a wing-body-horizontal-tail configuration designed for efficient performance at transonic speeds has been investigated at Mach numbers from 0.80 to 1.03 in the Langley 16-foot transonic tunnel. The effect of adding an outboard leading-edge chord-extension to the highly tapered 45 deg. swept wing was also obtained. The average Reynolds number for this investigation was 6.7 x 10(exp 6) based on the wing mean aerodynamic chord. The relatively low tail placement as well as the addition of a chord-extension achieved some alleviation of the pitchup tendencies of the wing-fuselage configuration. The maximum trimmed lift-drag ratio was 16.5 up to a Mach number of 0.9, with the moment center located at the quarter-chord point of the mean aerodynamic chord. For the untrimmed case, the maximum lift-drag ratio was approximately 19.5 up to a Mach number of 0.9.

Author

Body-Wing and Tail Configurations; Aircraft Design; Swept Wings; Wind Tunnel Tests; Transonic Speed; Cambered Wings; Aerodynamic Characteristics; Aerodynamic Balance

19980223615 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics at Mach Number 2.05 of a Series of Highly Swept Arrow Wings Employing Various Degrees of Twist and Camber

Carlson, Harry W., NASA Langley Research Center, USA; Oct. 1960; 36p; In English

Report No.(s): NASA-TM-X-332-1; L-876; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A series of arrow wings employing various degrees of twist and camber were tested in the Langley 4- by 4-foot supersonic pressure tunnel. Aerodynamic forces and moments in pitch were measured at a Mach number of 2.05 and at a Reynolds number of 4.4 x 10(exp 6) based on the mean aerodynamic chord. Three of the wings, having a leading-edge sweep angle of 70 deg. and an aspect ratio of 2.24, were designed to produce a minimum drag (in comparison with that produced for other wings in the family) at lift coefficients of 0, 0.08, and 0.16. A fourth and a fifth wing, having a 75 deg. swept leading edge and an aspect ratio of 1.65, were designed for lift coefficients of 0 and 0.16, respectively. A 70 deg. swept arrow wing with twist and camber designed for an optimum loading at a lift coefficient considerably less than that for maximum lift-drag ratio gave the highest lift-drag ratio of all the wings tested a value of 8.8 compared with a value of 8.1 for the corresponding wing without twist and camber. Two twisted and cambered wings designed for optimum loading at the lift coefficient for maximum lift-drag ratio gave only small increases in maximum lift-drag ratios over that obtained for the corresponding flat wings. However, in all cases, the lift-drag ratios obtained were far below the theoretical estimates.

Author

Aerodynamic Characteristics; Supersonic Speed; Arrow Wings; Swept Wings; Twisted Wings; Cambered Wings; Wind Tunnel Tests; Aerodynamic Coefficients; Aerodynamic Configurations

19980223947 NASA Langley Research Center, Hampton, VA USA

The Drag Coefficient of Parabolic Bodies of Revolution Operating at Zero Cavitation Number and Zero Angle of Yaw

Johnson, Virgil E., Jr., NASA Langley Research Center, USA; Rasnick, Thomas A., NASA Langley Research Center, USA; 1961; 20p; In English

Report No.(s): NASA-TR-R-86; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The form-drag coefficient of parabolic bodies of revolution with fineness ratios greater than 1 operating at zero angle of yaw and zero cavitation number is determined both theoretically and experimentally. Agreement between theory and experiment is very good. The theoretical form-drag coefficient of paraboloids is about half the form-drag coefficient of cones of comparable fineness ratio.

Author

Aerodynamic Drag; Bodies of Revolution; Parabolic Bodies; Aerodynamic Coefficients; Cavitation Flow

19980223966 NASA Langley Research Center, Hampton, VA USA

Transonic Aerodynamic Characteristics of a Model of a Proposed Six-Engine Hull-Type Seaplane Designed for Supersonic Flight

Wornom, Dewey E., NASA Langley Research Center, USA; Mar. 1960; 40p; In English

Report No.(s): NASA-TM-X-246; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Force tests of a model of a proposed six-engine hull-type seaplane were performed in the Langley 8-foot transonic pressure tunnel. The results of these tests have indicated that the model had a subsonic zero-lift drag coefficient of 0.0240 with the highest zero-lift drag coefficient slightly greater than twice the subsonic drag level. Pitchup tendencies were noted for subsonic Mach numbers at relatively high lift coefficients. Wing leading-edge droop increased the maximum lift-drag ratio approximately 8 percent at a Mach number of 0.80 but this effect was negligible at a Mach number of 0.90 and above. The configuration exhibited stable lateral characteristics over the test Mach number range.

Author

Transonic Speed; Aerodynamic Coefficients; Aerodynamic Characteristics; Seaplanes; Supersonic Flight; Wind Tunnel Tests; Wind Tunnel Models; Aerodynamic Configurations

19980223978 NASA Langley Research Center, Hampton, VA USA

Full-Scale Wind-Tunnel Investigation of the Drag Characteristics of an HU2K Helicopter Fuselage

Scallion, William L., NASA Langley Research Center, USA; 1963; 34p; In English

Report No.(s): NASA-TM-SX-848; L-3338; N-AM-110; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted in the Langley full-scale tunnel to determine the drag characteristics of the HU2K helicopter fuselage. The effects of body shape, engine operation, appendages, and leakage on the model drag were determined. The results of the tests showed that the largest single contribution to the parasite drag was that of the rotor hub installation which produced about 80 percent of the drag of the sealed and faired production body. Fairings on the rotor hub and blade retentions, or a cleaned-up hub and retentions, appeared to be the most effective single modifications tested. The total drag of all protuberances and air leakage also contributed a major part of the drag - an 83-percent increase over the drag of the sealed and faired production body. An additional increment of drag was caused by the basic shape of the fuselage - 19 percent more than the drag obtained when the fuselage shape was extensively refaired. Another sizable increment of drag was caused by the engine oil-cooler exit which gave a drag of 8 percent of that of the sealed and faired production body.

Author

Wind Tunnel Tests; Helicopters; Fuselages; Rotary Wings; Fairings; Hubs; Protuberances; Aerodynamic Drag

19980223979 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of the Pershing Missile During Separation of its Three Stages

McShera, John T., NASA Langley Research Center, USA; Townsend, Quwatha S., NASA Langley Research Center, USA; 1961; 94p; In English

Report No.(s): NASA-TM-SX-524; L-1360; A-AM-45; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An investigation to determine the aerodynamic characteristics of the first stage of a Pershing missile during separation from the second stage and of the second stage during separation from the reentry body has been made. The tests were conducted over a Mach number range from 1.70 to 4.65 and at Reynolds numbers of 3 x 10(exp 6) and 6 x 10(exp 6). The horizontal separation distance between the two stages was varied from 0 to 4 body diameters and the vertical separation distance was varied from 0 to 0.5 body diameters. The angle of attack of the forward stage was varied from 0 deg. to 6 deg., the angle of attack of stage one relative to the second stage was varied from 0 deg. to 5 deg., and the angle of attack of the second stage relative to the reentry body was varied from 0 deg. to 3 deg. Forces and moments were measured on the separated stage and the base pressure was measured on the forward stage. Tare forces caused by the presence of the support strut attached to the forward stage were approximated by making tests of the model inverted with and without an image strut. The results of this investigation are presented in coefficient form in tables without analysis.
Author
Aerodynamic Characteristics; Pershing Missile; Supersonic Speed; Missile Tests; Missile Trajectories

19980223980 **NASA Ames Research Center, Moffett Field, CA USA**
Investigation at Mach Numbers of 0.60 to 3.50 of Blended Wing-Body Combinations with Cambered and Twisted Wings with Diamond, Delta and Arrow Plan Forms
Holdaway, George H., NASA Ames Research Center, USA; Mellenthin, Jack A., NASA Ames Research Center, USA; Oct. 1960; 80p; In English
Report No.(s): NASA-TM-X-390; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This investigation is a continuation of the experimental and theoretical evaluation of blended wing-body combinations. The basic diamond, delta, and arrow plan forms which had an aspect ratio of 2 with leading-edge sweeps of 45.00 deg., 59.04 deg., and 70.82 deg. and trailing edge of -45.00 deg., -18.43 deg., and 41.19 deg., respectively, are used herein as standards for evaluating the effects of camber and warp. The wing thickness distributions were computed by varying the section shape along with the body radii (blending process) to match the prescribed area distribution and wing plan form. The wing camber and warp were computed to try to obtain nearly elliptical spanwise and chordwise load distributions for each plan form and thus to obtain low drag due to lift for a range of Mach numbers for which the velocities normal to the wing leading edge are subsonic. Elliptical chordwise load distributions were not possible for the plan forms and design conditions selected, so these distributions were somewhat different for each plan form. The models were tested with transition fixed at Mach numbers from 0.60 to 3.50 and at Reynolds numbers, based on the mean aerodynamic chord of the wing, of roughly 4,000,000 to 9,000,000. At speeds where the velocities normal to the wing leading edges were supersonic, an increase in the experimental wave-drag coefficients due to camber and twist was evident, but this penalty decreased with increased sweep. Thus the minimum wave-drag coefficients for the cambered arrow model were almost identical with the zero-lift wave- drag coefficients for the uncambered arrow model at all test Mach numbers.
Author
Body-Wing Configurations; Cambered Wings; Twisted Wings; Aerodynamic Coefficients; Airfoil Profiles; Subsonic Speed; Transonic Speed; Supersonic Speed

19980223994 **NASA Ames Research Center, Moffett Field, CA USA**
Large-Scale Wind-Tunnel Tests of a Wingless Vertical Take-Off and Landing Aircraft: Preliminary Results
Koenig, David G., NASA Ames Research Center, USA; Brady, James A., NASA Ames Research Center, USA; Oct. 1960; 44p; In English
Report No.(s): NASA-TN-D-326; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Large-scale wind-tunnel tests were made of a wingless vertical take-off and landing aircraft at zero sideslip to determine performance and longitudinal stability and control characteristics at airspeeds from 0 to 70 knots. Roll control and rudder effectiveness were also obtained. Limitations in the propulsion system restricted the lift for which level flight could be simulated to approximately 1500 pounds. Test variables with roll control and rudder undeflected were airspeed, vane setting, angle of attack, elevator deflection, and power. In most of the tests angle of attack, elevator, and power were varied individually while the other four parameters were held constant at previously determined values required for simulating trimmed level flight. The majority of the tests were made with power on and tail on at airspeeds between 20 and 70 knots. However, a limited number of data were obtained for the following conditions: (1) at zero velocity, horizontal tail on, power on; (2) at forward velocity, tail off and power on; and (3) at forward velocity, tail on, but with power off.
Author
Wind Tunnel Tests; Longitudinal Stability; Aerodynamic Characteristics; Aerodynamic Balance; Lateral Control; Vertical Landing; Aerodynamic Configurations; Wind Tunnel Models; Scale Models

19980223995 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics at Mach Numbers from to 1.6 to 2.8 of 74 deg. Swept Arrow Wings with and without Camber and Twist

Hasson, Dennis F., NASA Langley Research Center, USA; Fichter, Ann B., NASA Langley Research Center, USA; Wong, Norman, NASA Langley Research Center, USA; Sep. 1959; 38p; In English

Report No.(s): NASA-TM-X-8; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted to determine the lift, drag, and pitching-moment characteristics of a cambered and twisted arrow wing and an uncambered and untwisted arrow wing. The cambered and twisted wing was designed to give a high value of maximum lift-drag ratio at a lift coefficient of 0.1 and at a Mach number of 2.50. Each wing had a leading-edge sweep of 74 deg., an aspect ratio of 1.6, a taper ratio of 0, and a notch ratio of 0.714. A 3-percent-streamwise biconvex thickness distribution was centered on the mean camber surface of both wings. Tests were conducted at Mach numbers from 1.6 to 2.8 through a range of angle of attack from -6 deg. to 14 deg. The Reynolds number based on mean aerodynamic chord was 5.0 x 10(exp 6) for all tests. The maximum lift-drag ratio at the design Mach number for the cambered and twisted wing was 7.85 and, thus, was below the theoretically predicted value of 9.10. In addition, the cambered and twisted wing had only slightly higher values of maximum lift-drag ratio throughout the test Mach number range than the uncambered and untwisted wing. With the moment reference centers at 0.565 wing mean aerodynamic chord, both wings were slightly unstable longitudinally at low lift coefficients. For lift coefficients greater than about 0.1, the instability became more marked. These characteristics were obtained at all Mach numbers at which tests were made.

Author

Aerodynamic Characteristics; Aerodynamic Coefficients; Arrow Wings; Cambered Wings; Supersonic Speed; Swept Wings; Twisted Wings

19980227077 NASA Ames Research Center, Moffett Field, CA USA

Large-Scale Wind-Tunnel Tests of an Airplane Model with an Unswept, Tilt Wing of Aspect Ratio 5.5, and with Four Propellers and Blowing Flaps

Weiberg, James A., NASA Ames Research Center, USA; Holzhauser, Curt A., NASA Ames Research Center, USA; Jun. 1961; 34p; In English

Report No.(s): NASA-TN-D-1034; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were made of a large-scale tilt-wing deflected-slipstream VTOL airplane with blowing-type BLC trailing-edge flaps. The model was tested with flap deflections of 0 deg. without BLC, 50 deg. with and without BLC, and 80 deg. with BLC for wing-tilt angles of 0, 30, and 50 deg. Included are results of tests of the model equipped with a leading-edge flap and the results of tests of the model in the presence of a ground plane.

Author

Wind Tunnel Tests; Unswept Wings; Externally Blown Flaps; Boundary Layer Control; Aircraft Models; Trailing Edge Flaps; Vertical Takeoff Aircraft; Aerodynamic Coefficients; Tilt Wing Aircraft

19980227081 NASA Langley Research Center, Hampton, VA USA

Transonic Aerodynamic Loading Characteristics of a Wing-Body-Tail Combination Having a 52.5 deg. Sweptback Wing of Aspect Ratio 3 With Conical Wing Camber and Body Indentation for a Design Mach Number of Square Root of 2

Cassetti, Marlowe D., NASA Langley Research Center, USA; Re, Richard J., NASA Langley Research Center, USA; Igoe, William B., NASA Langley Research Center, USA; Oct. 1961; 102p; In English

Report No.(s): NASA-TN-D-971; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

An investigation has been made of the effects of conical wing camber and body indentation according to the supersonic area rule on the aerodynamic wing loading characteristics of a wing-body-tail configuration at transonic speeds. The wing aspect ratio was 3, taper ratio was 0.1, and quarter-chord-line sweepback was 52.5 deg. with 3-percent-thick airfoil sections. The tests were conducted in the Langley 16-foot transonic tunnel at Mach numbers from 0.80 to 1.05 and at angles of attack from 0 deg. to 14 deg., with Reynolds numbers based on mean aerodynamic chord varying from 7 x 10(exp 6) to 8 x 10(exp 6). Conical camber delayed wing-tip stall and reduced the severity of the accompanying longitudinal instability but did not appreciably affect the spanwise load distribution at angles of attack below tip stall. Body indentation reduced the transonic chordwise center-of-pressure travel from about 8 percent to 5 percent of the mean aerodynamic chord.

Author

Aerodynamic Loads; Body-Wing and Tail Configurations; Wind Tunnel Tests; Wing Loading; Sweptback Wings; Supersonic Speed; Conical Camber; Airfoil Profiles; Aerodynamic Characteristics

19980227085 NASA Langley Research Center, Hampton, VA USA

Upwash Characteristics at Several Stations on a Blunted Cone-Frustum-Cylinder Model at Mach Numbers from 1.60 to 4.65

Church, James D., NASA Langley Research Center, USA; Cremin, Joseph W., NASA Langley Research Center, USA; Nov. 1959; 26p; In English

Report No.(s): NASA-TM-X-59; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the upwash characteristics at several longitudinal stations along and above the surface of a blunted cone-frustion-cylinder model has been conducted in the Langley Unitary Plan wind tunnel. Data were obtained over a Mach number range from 1.60 to 4.65 at Reynolds numbers from approximately 2 x 10 (exp 6) to 4 x 10(exp 6) per foot depending on the Mach number. The data are presented as variations in upwash factor, defined as the slope of the local flow angle to the model angle of attack. Some of the effects of yaw angle, longitudinal station, and distance above the surface on the upwash factor are shown.

Author

Upwash; Supersonic Speed; Wind Tunnel Tests; Blunt Bodies; Cones; Frustums; Interference Drag; Cylindrical Bodies

19980227086 NASA Langley Research Center, Hampton, VA USA

Longitudinal Aerodynamic Characteristics of Ten Booster-Glider Models for Project Dyna-Soar at Mach Numbers of 0.7 and 1.0

West, F. E., Jr., NASA Langley Research Center, USA; Trescot, Charles D., Jr., NASA Langley Research Center, USA; Wiley, Alfred N., Jr., NASA Langley Research Center, USA; 1959; 36p; In English

Report No.(s): NASA-TM-SX-67; L-647; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been made of ten hypersonic booster-glider models at Mach numbers of 0-7 and 1.0. Only the booster portions of the model configurations were varied. Lift, drag, pitching-moment, and base-pressure data were obtained for a maximum angle-of-attack range of -10 to 11 deg. These data have not been analyzed since it was desired to expedite the publication of the results.

Author

Wind Tunnel Tests; Wind Tunnel Models; Hypersonic Gliders; Transonic Speed; Aerodynamic Characteristics; Booster Rocket Engines

19980227087 NASA Ames Research Center, Moffett Field, CA USA

An Investigation of the Pitching-Moment Contribution of a High Horizontal Tail on an Unswept-Wing and Body Combination at Mach Numbers from 0.80 to 1.40

Lippmann, Garth W., NASA Ames Research Center, USA; Aug. 1959; 32p; In English

Report No.(s): NASA-TM-X-43; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted to determine the effects of a high positioned horizontal tail on a wing-body configuration having a thin unswept wing of aspect ratio 3.09. Lift and pitching-moment coefficients were obtained for Mach numbers from 0.80 to 1.40 at Reynolds numbers of 1.0 and 1.5 million and for angles of attack to 20 deg. An experimental study of the pitching-moment contribution of the horizontal tail indicated that the marked destabilizing effect of the horizontal tail at high angles of attack for Mach numbers of 0.80 to 1.00 was associated with the formation of completely separated flow on the upper surface of the wing. Computations of the interference effects of the wing-body combination on the tail for Mach numbers of 0.80 and 0.94 and high angles of attack confirmed this conclusion. For a Mach number of 1.40, and high angles of attack, computations disclosed that the destabilizing effect primarily resulted from the trailing vortices of the wing. Two modifications to the basic wing plan form, which consisted of chord extensions, were generally unsuccessful in reducing the destabilizing contributions of the horizontal tail at high angles of attack.

Author

Transonic Speed; Unswept Wings; Body-Wing Configurations; Horizontal Tail Surfaces; Aerodynamic Coefficients; Separated Flow; Thin Wings; Lift; Maneuverability; Pitching Moments; Aircraft Configurations

19980227089 NASA Ames Research Center, Moffett Field, CA USA

Transition Reynolds Numbers of Separated Flows at Supersonic Speeds

Larson, Howard K., NASA Ames Research Center, USA; Keating, Stephen J., Jr., NASA Ames Research Center, USA; Dec. 1960; 32p; In English

Report No.(s): NASA-TN-D-349; A-178; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Experimental research has been conducted on the effects of wall cooling, Mach number, and unit Reynolds number on the transition Reynolds number of cylindrical separated boundary layers on an ogive-cylinder model. Results were obtained from

pressure and temperature measurements and shadowgraph observations. The maximum scope of measurements encompassed Mach numbers between 2.06 and 4.24, Reynolds numbers (based on length of separation) between 60,000 and 400,000, and ratios of wall temperature to adiabatic wall temperature between 0.35 and 1.0. Within the range of tile present tests, the transition Reynolds number was observed to decrease with increasing wall cooling, increase with increasing Mach number, and increase with increasing unit Reynolds number. The wall cooling effect was found to be four times as great when the attached boundary layer upstream of separation was cooled in conjunction with cooling of the separated boundary layer as when only the separated boundary layer was cooled. Wall cooling of both the attached and separated flow regions also caused, in some cases, reattachment in the otherwise separated region. Cavity resonance present in the separated region for some model configurations was accompanied by a large decrease in transition Reynolds number at the lower test Mach numbers.
Author
Boundary Layer Separation; Separated Flow; Walls; Cooling; Cylindrical Bodies; Ogives; Supersonic Speed; Wall Temperature; Reynolds Number

19980227094 NASA Langley Research Center, Hampton, VA USA
A Wind-Tunnel Investigation of the Development of Lift on Wings in Accelerated Longitudinal Motion
Turner, Thomas R., NASA Langley Research Center, USA; Aug. 1960; 18p; In English
Report No.(s): NASA-TN-D-422; L-1027; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was made in the Langley 300 MPH 7- by 10-foot tunnel to determine the development of lift on a wing during a simulated constant-acceleration catapult take-off. The investigation included models of a two-dimensional wing, an unswept wing having an aspect ratio of 6, a 35 deg. swept wing having an aspect ratio of 3.05, and a 60 deg. delta wing having an aspect ratio of 2.31. All the wings investigated developed at least 90 percent of their steady-state lift in the first 7 chord lengths of travel. The development of lift was essentially independent of the acceleration when based on chord lengths traveled, and was in qualitative agreement with theory.
Author
Lift; Delta Wings; Unswept Wings; Swept Wings; Wind Tunnel Tests; Aerodynamic Configurations; Aerodynamic Characteristics; Acceleration (Physics)

19980227096 NASA Ames Research Center, Moffett Field, CA USA
Large-Scale Wind-Tunnel Tests and Evaluation of the Low-Speed Performance of a 35 deg Sweptback Wing Jet Transport Model Equipped with a Blowing Boundary-Layer-Control Flap and Leading-Edge Slat
Hickey, David H., NASA Ames Research Center, USA; Aoyagi, Kiyoshi, NASA Ames Research Center, USA; Oct. 1960; 54p; In English
Report No.(s): NASA-TN-D-333; A-340; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A wind-tunnel investigation was conducted to determine the effect of trailing-edge flaps with blowing-type boundary-layer control and leading-edge slats on the low-speed performance of a large-scale jet transport model with four engines and a 35 deg. sweptback wing of aspect ratio 7. Two spanwise extents and several deflections of the trailing-edge flap were tested. Results were obtained with a normal leading-edge and with full-span leading-edge slats. Three-component longitudinal force and moment data and boundary-layer-control flow requirements are presented. The test results are analyzed in terms of possible improvements in low-speed performance. The effect on performance of the source of boundary-layer-control air flow is considered in the analysis.
Author
Boundary Layer Control; Sweptback Wings; Air Flow; Wind Tunnel Tests; Transport Aircraft; Blowing; Externally Blown Flaps; Aerodynamic Configurations; Trailing Edge Flaps; Jet Aircraft; Leading Edge Slats

19980227153 Naval Postgraduate School, Monterey, CA USA
Supersonic Flow Past Two Oscillating Airfoils
Alexandris, Georgios, Naval Postgraduate School, USA; Jun. 1998; 85p; In English
Report No.(s): AD-A350226; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Supersonic flow past two oscillating airfoils is analyzed using an elementary analytical theory valid for low frequencies of oscillation. The airfoils may have arbitrary stagger angle. This approach generalizes Sauer's solution for a single airfoil oscillating at small frequencies in an unbounded supersonic flow. It is shown that this generalization can provide an elementary theory for supersonic flow past two slowly oscillating airfoils. This aerodynamic tool will facilitate the evaluation of pressure distributions and consequently the calculation of moment coefficient. Torsional flutter boundaries are computed. The results for the pitch damping coefficient are the same when compared with previous analysis. For arbitrary frequencies a linearized method of characteristics was outlined. The elementary theory that has been developed in the thesis can be used for flutter evaluation of aircraft carrying

external stores. The result of the thesis is the derivation of the pitch-damping coefficient, which is necessary to predict the flutter conditions.
DTIC
Supersonic Flow; Airfoils; Oscillations

19980227178 NASA Langley Research Center, Hampton, VA USA
Free-Spinning-Tunnel Investigation of a 1/20-Scale Model of the North American T2J-1 Airplane
Bowman, James S., Jr., NASA Langley Research Center, USA; Healy, Frederick M., NASA Langley Research Center, USA; 1959; 22p; In English
Report No.(s): NASA-TM-SX-245; L-872; NASA-AD-3136; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made in the Langley 20-foot free-spinning tunnel to determine the erect and inverted spin and recovery characteristics of a 1/20-scale dynamic model of the North American T2J-1 airplane. The model results indicate that the optimum technique for recovery from erect spins of the airplane will be dependent on the distribution of the disposable load. The recommended recovery procedure for spins encountered at the flight design gross weight is simultaneous rudder reversal to against the spin and aileron movement to with the spin. With full wingtip tanks plus rocket installation and full internal fuel load, rudder reversal should be followed by a downward movement of the elevator. For the flight design gross weight plus partially full wingtip tanks, recovery should be attempted by simultaneous rudder reversal to against the spin, movement of ailerons to with the spin, and ejection of the wing-tip tanks. The optimum recovery technique for airplane-inverted spins is rudder reversal to against the spin with the stick maintained longitudinally and laterally neutral.
Author
Wind Tunnels; Wind Tunnel Stability Tests; Wind Tunnel Calibration; Scale Models; Loads (Forces)

19980227180 NASA Ames Research Center, Moffett Field, CA USA
Evaluation of Blended Wing-Body Combinations with Curved Plan Forms at Mach Numbers Up to 3.50
Holdaway, George H., NASA Ames Research Center, USA; Mellenthin, Jack A., NASA Ames Research Center, USA; Oct. 1960; 70p; In English
Report No.(s): NASA-TM-X-379; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

This investigation is a continuation of the experimental and theoretical evaluation of the effects of wing plan-form variations on the aerodynamic performance characteristics of blended wing-body combinations. The present report compares previously tested straight-edged delta and arrow models which have leading-edge sweeps of 59.04 and 70.82 deg., respectively, with related models which have plan forms with curved leading and trailing edges designed to result in the same average sweeps in each case. All the models were symmetrical, without camber, and were generally similar having the same span, length, and aspect ratios. The wing sections had an average value of maximum thickness ratio of about 4 percent of the local wing chords in a streamwise direction. The wing sections were computed by varying their shapes along with the body radii (blending process) to match the selected area distribution and the given plan form. The models were tested with transition fixed at Reynolds numbers of roughly 4,000,000 to 9,000,000, based on the mean aerodynamic chord of the wing. The characteristic effect of the wing curvature of the delta and arrow models was an increase at subsonic and transonic speeds in the lift-curve slopes which was partially reflected in increased maximum lift-drag ratios. Curved edges were not evaluated on a diamond plan form because a preliminary investigation indicated that the curvature considered would increase the supersonic zero-lift wave drag. However, after the test program was completed, a suitable modification for the diamond plan form was discovered. The analysis presented in the appendix indicates that large reductions in the zero-lift wave drag would be obtained at supersonic Mach numbers if the leading- and trailing-edge sweeps are made to differ by indenting the trailing edge and extending the root of the leading edge.
Author
Body-Wing Configurations; Supersonic Speed; Aerodynamic Characteristics; Wind Tunnel Tests; Planforms; Arrow Wings; Delta Wings

19980227195 NASA Dryden Flight Research Center, Edwards, CA USA
Preliminary Base Pressures Obtained from the X-15 Airplane at Mach Numbers from 1.1 to 3.2
Saltzman, Edwin J., NASA Dryden Flight Research Center, USA; Aug. 1961; 28p; In English
Report No.(s): NASA-TN-D-1056; H-215; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Base pressure measurements have been made on the fuselage, 10 deg.-wedge vertical fin, and side fairing of the X-15 airplane. Data are presented for Mach numbers between 1.1 and 3.2 for both powered and unpowered flight. Comparisons are made with data from small-scale-model tests, semiempirical estimates, and theory. The results of this preliminary study show that operation of the interim rocket engines (propellant flow rate approximately 70 lb/sec) reduces the base drag of the X-15 by 25 to 35 percent throughout the test Mach number range. Values of base drag coefficient for the side fairing and fuselage obtained from X-15 wind-tunnel models were adequate for predicting the overall full-scale performance of the test airplane. The leading-edge sweep of the upper movable vertical fin was not an important factor affecting the fin base pressure. The power-off base pressure coefficients of the upper movable vertical fin (a 10 deg. wedge with chord-to-thickness ratio of 5.5 and semispan-to-thickness ratio of 3.2) are in general agreement with the small-scale blunt-trailing-edge-wing data of several investigators and with two-dimensional theory.

Author

Base Pressure; Pressure Measurement; Pressure Ratio; Aerodynamic Coefficients; Transonic Speed; X-15 Aircraft

19980227199 NASA Langley Research Center, Hampton, VA USA

Investigation of Interference of a Deflected Jet with Free Stream and Ground on Aerodynamic Characteristics of a Semi-span Delta-Wing VTOL Model

Spreemann, Kenneth P., NASA Langley Research Center, USA; Aug. 1961; 92p; In English

Report No.(s): NASA-TN-D-915; L-1466; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An investigation of the mutual interference effects of the ground, wing, deflected jet stream, and free stream of a semispan delta-wing VTOL model at zero and low forward speeds has been conducted in the 17-foot test section of the Langley 300-MPH 7-by 10-foot tunnel. The model consisted of two interchangeable semispan clipped delta wings, a simplified fuselage, and a high-pressure jet for simulation of a jet exhaust. Attached to the wing behind the jet were various sets of vanes for deflecting the jet stream to different turning angles. The effect of ground proximity gave the normally expected losses in lift at zero and very low forward speeds (up to about 60 or 80 knots for the assumed wing loading of 100 lb/sq ft); at higher forward speeds ground effects were favorable. At low forward speeds, out of ground effect, the model encountered large losses in lift and large nose-up pitching moments with the model at low angles of attack and the jet deflected 90 deg or 75 deg (the angles required for VTOL performance and very low forward speeds). Rotating the model to higher angles of attack and deflecting the jet back to lower angles eliminated these losses in lift. Moving the jet rearward with respect to the wing reduced the losses in lift and the nose-up moments at all speeds within the range of this investigation.

Author

Jet Exhaust; Deflection; Delta Wings; Aerodynamic Characteristics; Angle of Attack

19980227207 NASA Ames Research Center, Moffett Field, CA USA

Axial-Force Reduction by Interference Between Jet and Neighboring Afterbody

Pitts, William C., NASA Ames Research Center, USA; Wiggins, Lyle E., NASA Ames Research Center, USA; Sep. 1960; 108p; In English

Report No.(s): NASA-TN-D-332; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

Experimental results are presented for an exploratory investigation of the effectiveness of interference between jet and afterbody in reducing the axial force on an afterbody with a neighboring jet. In addition to the interference axial force., measurements are presented of the interference normal force and the center of pressure of the interference normal force. The free-stream Mach number was 2.94, the jet-exit Mach number was 2.71, and the Reynolds number was 0.25 x 10, based on body diameter. The variables investigated include static-pressure ratio of the jet (up to 9), nacelle position relative to afterbody, angle of attack (-5 deg to 10 deg), and afterbody shape. Two families of afterbody shapes were tested. One family consisted of tangent-ogive bodies of revolution with varying length and base areas. The other family was formed by taking a planar slice off a circular cylinder with varying angle between the plane and cylinder. The trends with these variables are shown for conditions near maximum jet-afterbody interference. The interference axial forces are large and favorable. For several configurations the total afterbody axial force is reduced to zero by the interference.

Author

Afterbodies; Aerodynamic Interference; Jet Flow; Center of Pressure; Static Pressure; Pressure Ratio

19980227277 NASA Langley Research Center, Hampton, VA USA

A Transonic Investigation of Changing Indentation Design Mach Number on the Aeodynamic Characteristics of a 45 deg Sweptback-Wing-Body Combination Designed for High Performance

Loving, Donald L., NASA Langley Research Center, USA; Oct. 1961; 90p; In English

Report No.(s): NASA-TN-D-941; L-1698; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The effects of changing indentation design Mach number on the aerodynamic characteristics of a 45 deg. sweptback-wing-body combination designed for high performance have been investigated at Mach numbers from 0.80 to 1.13 in the Langley 8-foot transonic tunnel and at a Mach number of 1.43 in the Langley 8-foot transonic pressure tunnel. The Reynolds number of the inves-

tigation covered the range from approximately 2.5 x 10 (exp 6) to approximately 3.0 x 10(exp 6) based on the mean aerodynamic chord of the wing. The 45 deg. sweptback wing with camber and a thickened root was tested at 0 deg. angle of incidence on an unindented body and on bodies indented for Mach numbers M of 1.0, 1.2, and 1.4. Transonic and supersonic area rules were used in the design of the indented bodies. Theoretical zero-lift wave drag was calculated for these wing-body combinations. A -2 deg. angle of incidence of the wing, and M = 1.4 revised body indentation, and fixed transition also were investigated. Experimental values of zero-lift wave drag for the indented-body combinations followed closely the area-rule concept in that the lowest zero-lift wave-drag coefficient was obtained at or near the Mach number for which the body of the combination was designed. Theoretical values of zero-lift wave drag were considered to be in good agreement with the experimental results. At a given supersonic Mach number the highest values of maximum lift-drag ratio for the various combinations also were obtained at or near the Mach number for which the body of the combination was designed. At Mach numbers of 1.0, 1.2, and 1.43, the maximum lift-drag ratios were 15.3, 13.0, and 9.2, respectively. The use of an angle of incidence of -2 deg. for the wing in combination with the M = 1.2 body increased the zero-lift wave drag and decreased the maximum lift-drag ratio. All configurations maintained stable characteristics up to the highest lift coefficient of the investigation (C(L) approx. equal to 0.5).
Author
Aerodynamic Characteristics; Aerodynamic Coefficients; Sweptback Wings; Wind Tunnel Tests; Body-Wing Configurations; Transonic Speed

19980227281 **NASA Ames Research Center, Moffett Field, CA USA**
Theoretical Pressure Distributions on Wings of Finite Span at Zero Incidence for Mach Numbers Near 1
Alksne, Alberta Y., NASA Ames Research Center, USA; Spreiter, John R., NASA Ames Research Center, USA; 1961; 50p; In English
Report No.(s): NASA-TR-R-88; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method employed heretofore by the authors to obtain approximate solutions of the transonic flow equation for plane and axisymmetric flow is extended to give reasonable result for wings of finite span, consistent with the known properties of transonic flows. In this method the partial differential equation appropriate to the study of transonic flow is replaced by a nonlinear ordinary differential equation which can be solved by numerical methods. Asymptotic forms of this differential equation are given for very high and very low aspect ratios and analytic results are obtained for certain special cases. Numerical results, calculated by use of electronic computing machines, are given in the form of pressure distribution and pressure drag for two profile shapes, wedge and circular arc, for wings of rectangular plan form. The range of aspect ratios covered extends effectively from zero to infinity and agreement with asymptotic results is shown at both limits.
Author
Pressure Distribution; Wings; Pressure Drag; Axisymmetric Flow; Approximation; Numerical Analysis; Low Aspect Ratio; Transonic Flow

19980227287 **NASA Ames Research Center, Moffett Field, CA USA**
The Use of Drag Modulation to Limit the Rate at Which Deceleration Increases During Nonlifting Entry
Levy, Lionel L., Jr., NASA Ames Research Center, USA; Sep. 1961; 32p; In English
Report No.(s): NASA-TN-D-1037; A502; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The method developed in NASA TN D-319 for studying the atmosphere entry of vehicles with varying aerodynamic forces has been applied to obtain a closed-form solution for the motion, heating, range, and variation of the vehicle parameter m/C(D)A for nonlifting entries during which the rate of increase of deceleration is limited. The solution is applicable to vehicles of arbitrary weight, size, and shape, and to arbitrary atmospheres. Results have been obtained for entries into the earth's atmosphere at escape velocity during which the maximum deceleration and the rate at which deceleration increases were limited. A comparison of these results with those of NASA TN D-319, in which only the maximum deceleration was limited, indicates that for a given corridor depth, limiting the rate of increase of deceleration and the maximum deceleration requires an increase in the magnitude of the change in M/C(D)A and results in increases in maximum heating rate, total heat absorbed at the stagnation point, and range.
Author
Atmospheric Entry; Deceleration; Drag; Aerodynamic Forces; Aerospace Vehicles; Aeromaneuvering

19980227303 **NASA Ames Research Center, Moffett Field, CA USA**
Wind Tunnel Investigation at Supersonic Speeds of the Lift, Drag, Static-Stability and Control Characteristics of a 0.03-Scale Model of an Interim Development Version of the B-70 Airplane
Daugherty, James C., NASA Ames Research Center, USA; Green, Kendal H., NASA Ames Research Center, USA; Jun. 07, 1961; 154p; In English

Report No.(s): NASA-TM-SX-572; AF-AM-199; A-407; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

A 0.03-scale model of an interim development version of the B-70 airplane was tested at Mach numbers of 2.5, 3.0, and 3.5 at a Reynolds number of 5 million. Canard, elevon, aileron, and rudder control effectiveness were measured. A complete component drag evaluation study was made. Effects of modifications of the forebody boundary-layer gutter and inlet cowl-lip angle were obtained. In addition, the effects of increased wing camber were determined. Tests were conducted on the basic configuration with both natural and fixed boundary-layer transition. Trim drag estimates are made and the data are extrapolated to flight conditions at a Mach number of 3.0 to give maximum trimmed lift-drag ratios.

Author

Lift Drag Ratio; Boundary Layer Transition; Supersonic Speed; Wind Tunnel Tests; Aerodynamic Drag; Rudders; Ailerons

19980227305 **NASA Ames Research Center, Moffett Field, CA USA**

The Shock-Wave Patterns on a Cranked-Wing Configuration

Sammonds, Robert I., NASA Ames Research Center, USA; Nov. 1960; 16p; In English

Report No.(s): NASA-TN-D-346; A-433; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The shock-wave patterns of a complex configuration with cranked cruciform wings and a cone-cylinder body were examined to determine the interaction of the body bow wave with the flow field about the wing. Also of interest, was the interaction of the forward (760 sweptback) wing leading-edge wave with the rear (600 sweptback) wing leading-edge wave. The shadowgraph pictures of the model in free flight at a Mach number of 4.9, although not definitive, appear to indicate that the body bow wave crosses the outer wing panel after first being refracted either by the leading-edge wave of the 600 sweptback wing or by pressure fields in the flow crossing the wing.

Author

Shock Waves; Cruciform Wings; Bow Waves; Wing Planforms; Sweptback Wings; Aerodynamic Characteristics; Flow Distribution

19980227361 **NASA Langley Research Center, Hampton, VA USA**

Heat Transfer to 36.75 and 45 degree Swept Blunt Leading Edges in Free Flight at Mach Numbers from 1.70 to 2.99 and From 2.50 to 4.05

O'Neal, Robert L., NASA Langley Research Center, USA; Mar. 1960; 40p; In English

Report No.(s): NASA-TM-X-208; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation has been conducted to study the heat transfer to swept-wing leading edges. A rocket-powered model was used for the investigation and provided data for Mach number ranges of 1.78 to 2.99 and 2.50 to 4.05 with corresponding free-stream Reynolds number per foot ranges of 13.32 x 10(exp 6) to 19.90 x 10(exp 6) and 2.85 x 10(exp 6) to 4.55 x 10(exp 6). The leading edges employed were cylindrically blunted wedges ', three of which were swept 450 with leading-edge diameters of 1/4, 1/2, and 3/4 inch and one swept 36-750 with a leading-edge diameter of 1/2 inch. In the high Reynolds number range, measured values of heat transfer were found to be much higher than those predicted by laminar theory and at the larger values of leading-edge diameter were approaching the values predicted by turbulent theory. For the low Reynolds number range a comparison between measured and theoretical heat transfer showed that increasing the leading-edge diameter resulted in turbulent flow on the cylindrical portion of the leading edge.

Author

Heat Transfer; Swept Wings; Blunt Leading Edges; Free Flow; Turbulent Flow; Cylindrical Bodies

19980227409 **NASA Ames Research Center, Moffett Field, CA USA**

Longitudinal Force and Moment Data at Mach Numbers from 0.60 to 1.40 for a Family of Elliptic Cones with Various Semiapex Angles

Stivers, Louis S., Jr., NASA Ames Research Center, USA; Levy, Kionel L., Jr., NASA Ames Research Center, USA; Dec. 1961; 38p; In English

Report No.(s): NASA-TN-D-1149; A-548; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made to determine the aerodynamic characteristics of four elliptic cones having plan-form semiapex angles ranging from about 9 to 31 deg., and also for one of these cones modified on the upper surface to reduce the base area by about one half. The tests were made for angles of attack from about -2 to +21 deg., at Mach numbers from 0.60 to 1.40, and for a constant Reynolds number of 1.4 million, based on the length of the models. For each model, lift, pitching-moment, and drag coefficients, and lift-drag ratios are presented for the forebody, and axial-force coefficients are presented for the base. Calculated lift and pitching- moment curves for the elliptic cones, and lift-curve slopes for each model at supersonic Mach numbers are shown

for comparison with the corresponding experimental values. Lift-drag ratios are also given for the forebody and base combined. These data are presented without discussion.
Author
Aerodynamic Characteristics; Aerodynamic Coefficients; Aerodynamic Forces; Moment Distribution; Cones; Transonic Speed; Forebodies; Aerodynamic Configurations

19980227411 **NASA Langley Research Center, Hampton, VA USA**

Approximate Temperature Distributions and Streamwise Heat Conduction Effects in the Transient Aerodynamic Heating of Thin-Skinned Bodies

Conti, Raul J., NASA Langley Research Center, USA; Sep. 1961; 92p; In English
Report No.(s): NASA-TN-D-895; L-1227; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An approximate method is devised to determine temperature distributions during the transient aerodynamic heating of thin-skinned, heat-conducting bodies. This permits evaluation of the streamwise conduction errors arising in the measurement of heat-transfer coefficients based on the skin-temperature history. The present method is valid for a large range of body shapes and thickness distributions, within the limitations of one-dimensional (streamwise) heat conduction, quasi-isothermal surface, constant adiabatic wall temperature, and negligible radiative heat transfer. Numerical computations were carried out for flat plates, wedges, and conical, hemispherical, and hemicylindrical shells. The results are presented in the form of nondimensional charts that permit a rapid evaluation of a 10-percent error threshold in transient heat-transfer measurements.
Author
Conductive Heat Transfer; Aerodynamic Heating; Surface Temperature; Transient Heating; Conical Shells; Flat Plates; Wedges; Hemispherical Shells

19980227414 **NASA Ames Research Center, Moffett Field, CA USA**

Exploratory Study of the Reduction in Friction Drag Due to Streamwise Injection of Helium

Swenson, Byron L., NASA Ames Research Center, USA; Jan. 1961; 34p; In English
Report No.(s): NASA-TN-D-342; A-414; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The effects on average skin-friction drag and pressure drag of the streamwise injection of helium into the boundary layer near the nose of a 6 deg. half-angle cone at Mach numbers of 3 to 5 are presented. Large reductions in skin friction are shown to be possible with relatively small amounts of helium injection.
Author (revised)
Skin Friction; Pressure Drag; Half Cones; Wind Tunnel Tests; Gas Injection; Helium; Supersonic Speed; Friction Drag; Drag Reduction

19980227417 **NASA Langley Research Center, Hampton, VA USA**

Spin-Tunnel Investigation of a 1/20-Scale Model of the Northrop F-5E Airplane

Scher, Stanley H., NASA Langley Research Center, USA; White, William L., NASA Langley Research Center, USA; Sep. 1977; 48p; In English
Contract(s)/Grant(s): RTOP 505-11-41-08
Report No.(s): NASA-TM-SX-3556; L-11541; AF-AM-422; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley spin tunnel to determine the spin and recovery characteristics of a 1/20-scale model of the Northrop F-5E airplane. The investigation included erect and inverted spins, a range of center-of- gravity locations and moments of inertia, symmetric and asymmetric store loadings, and a determination of the parachute size required for emergency spin recovery. The effects of increased elevator trailing-edge-up deflections, of leading-edge and trailing-edge flap deflections, and of simulating the geometry of large external stores were also determined.
Author
F-5 Aircraft; Scale Models; Wind Tunnel Tests; Aircraft Spin; Spin Dynamics; Control Stability; Aircraft Control

19980227422 **Institute of Theoretical and Applied Mechanics, Novosibirsk, Russia**

International Conference on the Methods of Aerophysical Research 1998 "ICMAR 98", Part 1

1998; 259p; In English; Methods of Aerophysical Research, 29 Jun. - 3 Jul. 1998, Novosibirsk, Russia
Contract(s)/Grant(s): F61775-98-WE002
Report No.(s): AD-A350416; EOARD-CSP-98-1025; No Copyright; Avail: CASI; A12, Hardcopy; A03, Microfiche

The Final Proceedings for International Conference on Methods of Aerophysical Research (ICMAR'98), 29 June 1998 - 3 July 1998 This is an interdisciplinary conference. Topics include: Problems of Modeling at sub/trans/super/hypersonic velocities; Methods of flow diagnostics; Instrumentation for aerophysical experiments; Verification of CFD models and methods.
DTIC
Conferences; Computational Fluid Dynamics; Atmospheric Physics

19980227430 **NASA Langley Research Center, Hampton, VA USA**
Subsonic Aerodynamic Characteristics of the M117 Bomb with a Fragmentation Wrap
Capone, Francis J., NASA Langley Research Center, USA; Jul. 1965; 32p; In English
Report No.(s): NASA-TM-SX-1106; A-AM-77; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The aerodynamic effects of a fragmentation wrap around the cylindrical section of the M117 bomb have been determined in the Langley 16-foot transonic tunnel at Mach numbers from 0.30 to 0.90 and angles of attack from approximately -4 to 8 deg. The bomb without the wrap was also tested to a Mach number of 1.15. Total and static pressures measured at a fuze arming mechanism are also presented. The test Reynolds number based on a model length of 89.83 inches (228.17 cm) varied from 14.29 x 10(exp 6) to 29.99 x 10(exp 6).
Author
Aerodynamic Characteristics; Subsonic Speed; Bombs (Ordnance); Wind Tunnel Tests; Cylindrical Bodies; Aerodynamic Coefficients

19980227736 **NASA Langley Research Center, Hampton, VA USA**
Study of Flow Over Oscillating Airfoil Models at a Mach Number of 7.0 in Helium
Arman, Ali, NASA Langley Research Center, USA; Dec. 1961; 24p; In English
Report No.(s): NASA-TN-D-992; L-839; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel study of unsteady flow at a Mach number of 7 in helium has been conducted on several sting-mounted wedge, double-wedge, and flat-plate airfoil models with three different leading-edge radii. The data were obtained by taking high-speed schlieren motion pictures of the decaying motion of the model as it was released from an initial deflection. The shock-wave position observed on the sharp-leading-edge models during the oscillation was compared with that obtained by use of unsteady flow theory as well as steady-state theory. Comparison of theoretical results indicated that no unsteady-flow effects exist over the range of reduced frequencies k, 0.007 less than equal than k less than or equal 0.030, studied experimentally. The experimental results confirmed this finding as no unsteady-flow effects were detected in this reduced-frequency range. Comparison of shock-wave positions measured for the blunt models with those calculated by steady-state methods indicated fair agreement.
Author
Unsteady Flow; Wind Tunnel Tests; Airfoil Oscillations; Aeroelasticity; Hypersonic Speed; Flat Plates; Leading Edges

19980227752 **NASA Langley Research Center, Hampton, VA USA**
Equations for the Induced Velocities Near a Lifting Rotor with Nonuniform Azimuthwise Vorticity Distribution
Heyson, Harry H., NASA Langley Research Center, USA; Aug. 1960; 28p; In English
Report No.(s): NASA-TN-D-394; L-797; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Equations, which can be integrated on high-speed computing machines, are developed for all three components of induced velocity at an arbitrary point near the rotor and for an arbitrary harmonic variation of vorticity. Sample calculations for vorticity which varies as the sine of the azimuth angle indicate that the normal component of induced velocity is, in this case, uniform along either side of the lateral axis.
Author
Rotor Aerodynamics; Lifting Rotors; Vorticity; Flow Distribution; Flow Velocity; Velocity Distribution; Aerodynamic Interference

19980227753 **NASA Langley Research Center, Hampton, VA USA**
Free-Spinning-Tunnel Investigation of a 1/20-Scale Model of an Unswept-Wing Jet-Propelled Trainer Airplane
Bowman, James S., Jr., NASA Langley Research Center, USA; Healy, Frederick M., NASA Langley Research Center, USA; Jun. 1960; 20p; In English
Report No.(s): NASA-TN-D-381; L-872; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flutter analysis employing the kernel function for three- dimensional, subsonic, compressible flow is applied to a flutter-tested tail surface which has an aspect ratio of 3.5, a taper ratio of 0.15, and a leading-edge sweep of 30 deg. Theoretical and experimental results are compared at Mach numbers from 0.75 to 0.98. Good agreement between theoretical and experimental flutter

dynamic pressures and frequencies is achieved at Mach numbers to 0.92. At Mach numbers from 0.92 to 0.98, however, a second solution to the flutter determinant results in a spurious theoretical flutter boundary which is at a much lower dynamic pressure and at a much higher frequency than the experimental boundary.
Author
Wind Tunnel Tests; Unswept Wings; Aircraft Spin; Subsonic Flow; Scale Models; Flutter Analysis; Spin Dynamics

19980227791 NASA Ames Research Center, Moffett Field, CA USA
The Numerical Calculation of Flow Past Conical Bodies Supporting Elliptic Conical Shock Waves at Finite Angles of Incidence
Briggs, Benjamin R., NASA Ames Research Center, USA; Nov. 1960; 68p; In English
Report No.(s): NASA-TN-D-340; A-385; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The inverse method, with the shock wave prescribed to be an elliptic cone at a finite angle of incidence, is applied to calculate numerically the supersonic perfect-gas flow past conical bodies not having axial symmetry. Two formulations of the problem are employed, one using a pair of stream functions and the other involving entropy and components of velocity. A number of solutions are presented, illustrating the numerical methods employed, and showing the effects of moderate variation of the initial parameters.
Author
Conical Bodies; Shock Waves; Supersonic Flow; Aerodynamic Configurations; Ideal Gas

19980227792 NASA Langley Research Center, Hampton, VA USA
Transonic Wind-Tunnel Investigation of the Fin Loads on a 1/8-Scale Model Simulating the First Stage of the Scout Research Vehicle
Kelly, Thomas C., NASA Langley Research Center, USA; Jun. 1961; 50p; In English
Report No.(s): NASA-TN-D-918; L-1438; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation to determine the fin loads on a 1/8-scale model simulating the first stage of the Scout research vehicle was made in the Langley 8-foot transonic tunnel at Mach numbers from 0.40 to 1.20. Tests were conducted over an angle-of-attack range from about -10 to 10 deg and at a Reynolds number per foot of approximately 3.5 x 10(exp 6). Results of the tests indicate that for a given angle of attack, negative tip-control deflections caused decreases in normal-force and fin-bending-moment coefficients and increases in pitching-moment coefficient, as would be expected. The effects were slight at a model angle of attack of -10 deg where tip-control stall had probably occurred but increased with an increase in angle of attack.
Author
Aerodynamic Loads; Wind Tunnel Tests; Transonic Speed; Fins; Research Vehicles; Scale Models; Aerodynamic Coefficients

19980227793 NASA Langley Research Center, Hampton, VA USA
Wind-Tunnel Investigation of a Balloon as a Towed Decelerator at Mach Numbers from 1.47 to 2.50
McShera, John T., NASA Langley Research Center, USA; Keyes, J. Wayne, NASA Langley Research Center, USA; Aug. 1961; 82p; In English
Report No.(s): NASA-TN-D-919; L-884; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been conducted to study the characteristics of a towed spherical balloon as a drag device at Mach numbers from 1.47 to 2.50, Reynolds numbers from 0.36 x 10(exp 6) to 1.0 x 10(exp 6) , and angles of attack from -15 to 15 deg. Towed spherical balloons were found to be stable at supersonic speeds. The drag coefficient of the balloon is reduced by the presence of a tow cable and a further reduction occurs with the addition of a payload. The balloon inflation pressure required to maintain an almost spherical shape is about equal to the free-stream dynamic pressure. Measured pressure and temperature distribution around the balloon alone were in fair agreement with predicted values. There was a pronounced decrease in the pressure coefficients on the balloon when attached to a tow cable behind a payload.
Author
Balloons; Supersonic Speed; Towed Bodies; Wind Tunnel Tests; Aerodynamic Coefficients; Free Flow

19980227802 NASA Langley Research Center, Hampton, VA USA
Aerodynamic Characteristics, Temperature, and Noise Measurements of a Large-Scale External-Flow Jet-Augmented-Flap Model with Turbojet Engines Operating
Fink, Marvin P., NASA Langley Research Center, USA; Sep. 1961; 50p; In English
Report No.(s): NASA-TN-D-943; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley full-scale tunnel on a large-scale model powered by turbojet engines with flattened rectangular nozzles. The wing had 35 deg. sweep of the leading edge, an aspect ratio of 6.5, a taper ratio of 0.31, and NACA 65(1)-412 and 65-408 airfoils at the root and tip. The investigation included measurements of the longitudinal aerodynamic characteristics of the model with half-span and full-span flaps and measurements of the sound pressure and skin temperature on the portions of the lower surface of the wing immersed in the jet flow. The tests were conducted over a range or angles of attack from -8 to 16 deg. for Reynolds numbers from 1.8 x 10(exp 6) to 4.4 x 10(exp 6) and a range of momentum coefficients from 0 to 2.0. In general, the aerodynamic results of this investigation made with a large-scale hot-jet model verified the results of previous investigations with small models powered by compressed-air jets. Although blowing was only done over the inboard portion of the wing, substantial amounts of induced lift were also obtained over the outboard portion of the wing. Skin temperatures were about 340 F and wing heating could be handled with available materials without cooling. Random acoustic loadings on the wing surface were high enough to indicate that fatigue failure from this source would require special consideration in the design of an external-flow jet flap system for an airplane.

Author

Aerodynamic Characteristics; Noise Measurement; Temperature; Turbojet Engines; Airfoils; Wind Tunnel Models; Scale Models; Wind Tunnel Tests; Lift Augmentation; Sweptback Wings; Jet Flow

19980227803 **NASA Langley Research Center, Hampton, VA USA**

Experimental and Theoretical Deflections and Natural Frequencies of an Inflatable Fabric Plate

Stroud, W. Jefferson, NASA Langley Research Center, USA; Oct. 1961; 32p; In English

Report No.(s): NASA-TN-D-931; L-1317; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Static and vibration tests were performed on an inflatable square fabric plate supported on all edges. Lateral deflections and natural frequencies showed good agreement with calculations made using a linear small-deflection theory.

Author

Structural Analysis; Inflatable Structures; Static Tests; Vibration Tests; Plates (Structural Members); Fabrics; Deflection; Resonant Frequencies

03
AIR TRANSPORTATION AND SAFETY

Includes passenger and cargo air transport operations; and aircraft accidents.

19980221785 **European Organization for the Safety of Air Navigation, Bretigny-sur-Orge, France**

ATFM Studies: Remaining Overdeliveries

Ganvert, E., European Organization for the Safety of Air Navigation, France; Greiling, Y., European Organization for the Safety of Air Navigation, France; Vidal, A., European Organization for the Safety of Air Navigation, France; Mar. 1998; 26p; In English

Report No.(s): PB98-164304; EEC/NOTE-5/98; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This document describes an ATFM (Air Traffic Flow Management) study conducted by the Centre of Expertise Flight Data Research on behalf of the CFMU (Central Flow Management Unit) in to evaluate the performance of the current CFMU Operations and to evaluate the Slot Allocation process by analyzing the remaining overdeliveries on regulated sectors.

NTIS

Air Traffic Control; Airports; Flight Management Systems; Flow Distribution

19980221788 **European Organization for the Safety of Air Navigation, Experimental Centre, Bretigny-sur-Orge, France**

Coverage of European Air Traffic for the Base Aircraft Data (BADA)

Bos, A., European Organization for the Safety of Air Navigation, France; Mar. 1998; 30p; In English

Report No.(s): PB98-164320; EEC/NOTE-8/98-Rev-3.0; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The air traffic statistics from the CFMU for December 1997 and January 1998 are used to determine the coverage of European air traffic by the Base of Aircraft Data (BADA) Revision 3.0 BADA consists of a set of aircraft models used at the EEC and other European research institutes for aircraft trajectory simulation. The results show that the 67 aircraft types within BADA 3.0 cover 89.4% of the European air traffic. The addition of 1 type would bring the coverage to the target of 90%.

NTIS

Air Traffic; Research Aircraft; Aircraft Models

19980221792 Federal Aviation Administration, Washington, DC USA
Notices to Airmen: Domestic/International
Jul. 16, 1998; 238p; In English
Report No.(s): PB98-163389; No Copyright; Avail: CASI; A11, Hardcopy; A03, Microfiche

Table of Contents: Airway Notams; Airports, Facilities, and Procedural Notams; General FDC Notams; Part 95 Revisions to Minimum En Route IFR Altitudes and Changeover Points; International Notices to Airmen; and Graphic Notices.

NTIS

National Airspace System; Air Navigation; Airports; Altitude; Graphs (Charts); Constrictions

19980221806 Federal Aviation Administration, FAA Technical Center, Atlantic City, NJ USA
Functional Requirements for Screener Assist Technologies
Fobes, J. L., Federal Aviation Administration, USA; Neiderman, Eric C., Federal Aviation Administration, USA; Jul. 1998; 36p; In English
Report No.(s): PB98-159742; DOT/FAA/AR-98/35; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This document lists the human factors functional requirements for Screener Assist Technologies (SAT) to enhance screener performance to detect threat objects. The report also describes the required interactions with Threat Image Projection (TIP) systems, naming conventions for threats, data report capabilities, FAA acceptance test procedures, and operational and technical criteria that will be used to assess system effectiveness.

NTIS

Functional Design Specifications; Human Factors Engineering; Aircraft Safety; Airport Security; X Ray Inspection

19980223924 Texas Univ., Health Science Center, Houston, TX USA
General Aviation Accidents: The USA Air Force Aero Club Solution
Brandt, Keith E., Texas Univ., USA; Aug. 07, 1998; 72p; In English
Report No.(s): AD-A350974; AFIT-98-049; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Aviation is an intrinsically safe mode of travel. In 1994, the United States Air force system of Aero Clubs put forth substantial effort to put a program in place (fly Smart) to improve flying safety in its aircraft. This study compares the accident rates of Aero Club aircraft with rates seen in general aviation. A comparison is also made of the years prior to implementation of fly Smart to the three years following implementation. Aero Club records of accidents were available from 1987 through 1997. General aviation mishap statistics are collected by the National Transportation Safety Board and are collected and presented to the public by the Aircraft Owners and Pilots Association in the form of an annual general aviation report. Comparison of these figures show that the Aero Club system had a lower accident rate and fatality rate in all but one study year (1992, Aero Club 10.12 accidents and 2.38 fatal accidents per 100,000 flying hours; general aviation 8.97 accidents and 1.75 fatal accidents per 100,000 flying hours). The Aero Club accident rate in the period following implementation of fly Smart (1995 - 1997) was lower than before implementation (1987 - 1993, 5.19 versus 1.63, $p=0.047$), while general aviation rates for the same periods were unchanged (8.29 versus 8.00, $p>0.05$). No differences were seen in rates of larger vs. mid-size or small clubs. There were no differences in the accident rates of closed vs. open clubs. The Air Force Aero Clubs are certainly more restrictive than general aviation, but the improvement in safety record suggests the tighter regulations are rules you can live with.

DTIC

Aircraft Accidents; Flight Safety; Safety Management; Civil Aviation

19980227159 Naval Postgraduate School, Monterey, CA USA
Allocating Flight Hours to Army Helicopters
Pippin, Bradley W., Naval Postgraduate School, USA; Jun. 1998; 55p; In English
Report No.(s): AD-A350138; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Army helicopter battalions, consisting of 24 helicopters valued from $206.4 million (UH-60 Blackhawk battalion) to $432 million (AH-64 Apache battalion), allocate flight hours to helicopters using manual techniques that have caused an unnecessary decrease in battalion deployability. This thesis models the battalion's flight hour allocation problem using optimization; it develops both a mixed integer linear program and a quadratic program. The 2nd Battalion, 4th Aviation Regiment of 4th Mechanized Division currently uses a spreadsheet implementation of the quadratic program developed by the author called QFHAM (Quadratic Flight Hour Allocation Model), that is available to other battalions for use with existing software and computer resources. The mixed integer linear program, called FHAM (Flight Hour Allocation Model) more appropriately models the problem, but requires additional software. This thesis validates the two models using actual flight hour data from a UH-60 battalion under both typical training and contingency scenarios. The models provide a monthly flight hour allocation for the battalion's aircraft that

results in a steady-state sequencing of aircraft into phase maintenance, thus eliminating phase maintenance backlog and providing a fixed number of aircraft available for deployment. This thesis also addresses the negative impact of current helicopter battalion readiness measures on deployment and offers alternatives.
DTIC
Scheduling; Allocations; AH-64 Helicopter; Computer Systems Programs

19980227419 NASA Lewis Research Center, Cleveland, OH USA
A Combined Water-Bromotrifluoromethane Crash-Fire Protection System for a T-56 Turbopropeller Engine
Campbell, John A., NASA Lewis Research Center, USA; Busch, Arthur M., NASA Lewis Research Center, USA; Aug. 1959; 36p; In English
Report No.(s): NASA-TN-D-28; E-308; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A crash-fire protection system is described which will suppress the ignition of crash-spilled fuel that may be ingested by a T-56 turbo-propeller engine. This system includes means for rapidly extinguishing the combustor flame, means for cooling and inerting with water the hot engine parts likely to ignite engine ingested fuel, and means for blanketing with bromotrifluoromethane massive metal parts that may reheat after the engine stops rotating. Combustion-chamber flames were rapidly extinguished at the engine fuel nozzles by a fuel shutoff and drain valve. Hot engine parts were inerted and cooled by 42 pounds of water discharged at seven engine stations. Massive metal parts that could reheat were inerted with 10 pounds of bromotrifluoromethane discharged at two engine stations. Performance trials of the crash-fire protection system were conducted by bringing the engine up to takeoff temperature, actuating the crash-fire protection system, and then spraying fuel into the engine to simulate crash-ingested fuel. No fires occurred during these trials, although fuel was sprayed into the engine from 0.3 second to 15 minutes after actuating the crash-fire protection system.
Author
Fire Prevention; Engine Parts; T-56 Engine; Extinguishing; Fire Extinguishers; Crashes; Aircraft Fuels; Spilling; Spraying; Water

04
AIRCRAFT COMMUNICATIONS AND NAVIGATION

Includes digital and voice communication with aircraft; air navigation systems (satellite and ground based); and air traffic control.

19980223080 Rockwell Collins, Inc., Advanced Technology Center, Cedar Rapids, IA USA
Integrated Airport Surface Operations
Koczo, S., Rockwell Collins, Inc., USA; Jul. 1998; 180p; In English
Contract(s)/Grant(s): NAS1-19704; RTOP 538-04-13-02
Report No.(s): NASA/CR-1998-208441; NAS 1.26:208441; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

The current air traffic environment in airport terminal areas experiences substantial delays when weather conditions deteriorate to Instrument Meteorological Conditions (IMC). Research activity at NASA has culminated in the development, flight test and demonstration of a prototype Low Visibility Landing and Surface Operations (LVLASO) system. A NASA led industry team and the FAA developed the system which integrated airport surface surveillance systems, aeronautical data links, DGPS navigation, automation systems, and controller and flight deck displays. The LVLASO system was demonstrated at the Hartsfield-Atlanta International Airport using a Boeing 757-200 aircraft during August, 1997. This report documents the contractors role in this testing particularly in the area of data link and DGPS navigation.
Author
Air Traffic Control; Data Links; All-Weather Landing Systems; Boeing 757 Aircraft; Instrument Flight Rules; All-Weather Air Navigation; Navigation Instruments

19980227146 Federal Aviation Administration, Technical Center, Atlantic City, NJ USA
Traffic Information Service (TIS) Developmental/Operational Test and Evaluation (DT/E and OT/E) *Final Report*
McNeil, Michael, Federal Aviation Administration, USA; Sharkey, Robert, Federal Aviation Administration, USA; Jun. 1998; 145p; In English
Report No.(s): AD-A350376; DOT/FAA/CT-TN98/10; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

The Federal Aviation Administration (FAA) Traffic Information Service (TIS) Developmental Test and Evaluation (DT&E) and Operational Test and Evaluation (OT&E) Final Test Report is prepared by the Mode Select (Mode S) Test Group of the Surveillance Branch ACT-310. It provides the detailed analysis, results, the final conclusions, and recommendations drawn from the

DT&E and OT&E of the TIS data link service for the Mode S Beacon Radar System. The purpose of the TIS data link function is intended to improve the safety and efficiency of "see-and-avoid" flight by providing automatic display to the pilot of nearby traffic and warnings of any potentially threatening conditions. The source of TIS information is the file of aircraft tracks maintained by the ground Mode S sensor providing coverage for a region of airspace.

DTIC

Air Traffic; Systems Analysis; Data Links; Beacons; Airspace

19980227148 Federal Aviation Administration, Technical Center, Atlantic City, NJ USA

Reduced Horizontal Separation Minima (RHSM) Concept Exploration Simulation

Elkan, Elizabeth, Federal Aviation Administration, USA; Kopardekar, Parimal, Federal Aviation Administration, USA; Stahl, David, Federal Aviation Administration, USA; Mar. 1998; 44p; In English

Report No.(s): AD-A350324; DOT/FAA/CT-TN97/3; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The Informal South Pacific Air Traffic Services Coordinating Group has been investigating a number of concepts to improve operational efficiency for flights in the Pacific Oceanic region. The Federal Aviation Administration (FAA) Air Traffic Requirements (ATR-3 10) and Air Traffic Operations (ATO-100) program offices tasked the Simulation and Systems Integration Branch (ACT-540), in cooperation with the Oceanic and Offshore Integrated Product Team (AUA-600), to explore the feasibility of implementing reduced oceanic aircraft separations. These organizations formed an Experimental Working Group to make high-level decisions regarding the implementation of the proposed separation standard. In response, ACT-540 formed a Research Team to design and conduct a concept exploration study at the FAA William J. Hughes Technical Center. The Research Team led all efforts including the planning and design of the simulation and conduct of a simulation. The team also queried the controllers and compiled their responses regarding the proposed procedure. This report discusses the Reduced Horizontal Separation Minima (RHSM) concept exploration simulation. It describes the simulation, procedures, and tools developed to ascertain the experiences of individuals who participated. The concept exploration examined issues that might affect a controller's ability to manage reduced longitudinal separation in the oceanic environment. A demonstration of the RHSM concept was conducted in the Oceanic Laboratory at the Federal Aviation Administration (FAA) William J. Hughes Technical Center on November 6 and 7, 1996.

DTIC

Air Traffic; Controllers; Pacific Ocean

19980227311 Federal Aviation Administration, Civil Aeromedical Inst., Oklahoma City, OK USA

The Relationship of Sector Characteristics to Operational Errors *Final Report*

Rodgers, Mark D.; Mogford, Richard H.; Mogford, Leslye S.; May 1998; 66p; In English; Prepared in collaboration with William J. Hughes Technical Center, Atlantic City, NJ and Rigel Associates, Marmora, NJ.

Contract(s)/Grant(s): DTFA02-95-P-35434

Report No.(s): AD-A350717; DOT/FAA/AM-98/14; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An exploratory study was conducted on the relationship of air traffic control (ATC) complexity factors to operational errors (OEs). This consisted of a detailed examination of OE data from 1992 through 1995 from the Atlanta en route center. The Systematic Air Traffic Operations Research Initiative (SATORI) system was used to collect data for the analysis. Sectors were categorized into zero-, low-, and high-error groups. Fifteen sector and traffic flow variables had statistically significant correlations with OE frequency. Four variables were higher for the high-error group as compared to the zero-error group. Sector size was smaller for the high-error group as compared to the combined zero- and low-error categories. A significant multiple correlation was found between overall OE rate and a subset of the ATC complexity measures. The data were also analyzed to define relationships between the complexity measures and controller situational awareness (SA) at the time of the OE. The only statistically significant difference between OEs with and without SA was for horizontal separation. In addition, high-error sectors were characterized by low SA for errors. Certain sector and traffic flow characteristics were associated with these high-error sectors, suggesting that these factors may negatively affect SA. It was concluded that the results demonstrated a relationship between sector complexity and OE rate. Such findings, if extended, could assist with traffic management, sector design activities, and the development of decision-support systems.

DTIC

Operations Research; Errors; Air Traffic Control; Error Analysis

19980227320 Oklahoma Univ., Dept. of Psychology, Norman, OK USA

Aircraft Importance and Its Relevance to Situation Awareness *Final Report*

Gronlund, Scott D.; Ohrt, Daryl D.; Dougherty, Michael R.; Perry, Jennifer L.; Manning, Carol A.; May 1998; 14p; In English

Contract(s)/Grant(s): DTFA02-93-D-93088

Report No.(s): AD-A350417; DOT/FAA/AM-98/16; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

We tested en route air traffic controllers (currently serving as instructors at the FAA Academy) to determine what they remember about the aircraft in their sector. We focused on memory for flight data (especially aircraft altitude and ground speed) and the position of the aircraft on the radar screen. Aircraft importance affected memory for flight data but not the highly accurate recall of the radar position of the aircraft. We hypothesize that controllers use their excellent memory for aircraft position to classify aircraft as important (potential traffic) or not, and better remember flight data about important aircraft (in particular, their exact altitude). The results have implications for improving techniques to assess situation awareness and interfaces to support it.

DTIC

Air Traffic Controllers (Personnel); Alertness; Memory; Air Traffic Control

19980227343 Civil Aeromedical Inst., Oklahoma City, OK USA

The Combination of Flight Count and Control Time as a New Metric of Air Traffic Control Activity *Final Report*

Mills, Scott H.; May 1998; 15p; In English

Report No.(s): AD-A350504; DOT/FAA/AM-98/15; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The exploration of measures of airspace activity is useful in a number of significant ways, including the establishment of baseline air traffic control (ATC) measures and the development of tools and procedures for airspace management. This report introduces a new metric of ATC activity that combines two existing measures (flight count and the time aircraft are under control). The Aircraft Activity Index (AAI) is sensitive to changes in both flight count and flight length, and therefore is a superior measure for comparing aircraft activity between two epochs of time. The AAI was applied to data from 10 days of System Analysis Recordings obtained from the Seattle Air Route Control Center. The advantages of the AAI were most apparent when different aircraft types consistently had different mean flight lengths. Possible uses of the AAI and other ATC measures for the evaluation of new systems and procedures are discussed.

DTIC

Air Traffic Control; Systems Analysis; Airspace

19980227346 Mississippi State Univ., Aerospace Engineering, Mississippi State, MS USA

Flight Test Evaluation of a Differential Global Positioning System Sensor in Runway Performance Testing

Germann, Kenneth Paul; Aug. 04, 1998; 78p; In English

Report No.(s): AD-A350715; 98-035; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This study discusses the use of a carrier phase differential global positioning system (DGPS) receiver set in basic takeoff and landing performance flight testing. A technique for using DGPS receivers as theodolites in takeoff and landing performance tests is developed. Both position and velocity data are available from a DGPS receiver. As a result distances can be calculated by differencing the position coordinates or by integrating the available ground velocities. Both of these techniques are used and compared to a traditional video theodolite system for ground roll distances. . The viability of using DGPS ground speed data in lieu of air data in calculating the distance to clear a barrier is also explored. These methods are used to determine the nominal takeoff and landing performance of an experimental general aviation airplane. Test results are mixed. DGPS velocity integration yields good results for ground phase calculations. All other results are inconclusive.

DTIC

Flight Tests; Global Positioning System; Runways; Performance Tests; Receivers; Takeoff; Aircraft Landing; Aircraft Performance

19980227424 Civil Aeromedical Inst., Oklahoma City, OK USA

An Analysis of Voice Communication in a Simulated Approach Control Environment *Final Report*

Prinzo, O. V., Civil Aeromedical Inst., USA; May 1998; 30p; In English

Report No.(s): AD-A350523; DOT/FAA/AM-98/17; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report consists of an analysis of simulated terminal radar approach control (TRACON) air traffic control communications. Twenty-four full performance level air traffic controllers (FPLATC) from 2 TRACON facilities participated in the simulation study. Each controller worked 2 light- and 2 heavy-traffic density scenarios for feeder and final sectors. All communications were audio recorded and transcribed verbatim by a retired FPLATC. Once transcribed, transmissions were parsed into communication elements. Each communication element was assigned a speech act category (e.g., address, instruction, request, or advisory), an aviation topic (e.g., altitude, heading, speed) and then coded for irregularities (e.g., grouping numbers together when they should be spoken sequentially, or omitting, substituting, or adding words contrary to required phraseology) (ATSAT, Prinzo et al., 1995). The simulated communications were compared to an analysis performed on audiotapes from the same TRACON facilities. Percentages in 3 speech act categories were comparable (Instruction, 55% versus 51%; Address; 14% versus 26%; Advi-

sory, 24% versus 18%). Detailed analyses revealed that, although there were fewer irregular communications produced during simulation, the distributions of those communication irregularities were very much the same, with the exception of aircraft call sign. The differences in those distributions were attributed to the voice recognition system; it could not recognize a call sign spoken sequentially and then restated in grouped form.
DTIC
Voice Communication, Simulation; Controllers; Radar Approach Control; Air Traffic Controllers (Personnel); Air Traffic Control

05
AIRCRAFT DESIGN, TESTING AND PERFORMANCE

Includes aircraft simulation technology

19980221786 **Federal Aviation Administration, Fire Safety Section, Atlantic City, NJ USA**
Cargo Compartment Fire Protection in Large Commercial Tran[illegible] Aircraft
Blake, D., Federal Aviation Administration, USA; Marker, T., Federal Aviation Administration, USA; Hill, R., Federal Aviation Administration, USA; Reinhardt, J., Federal Aviation Administration, USA; Sarkos, C., Federal Aviation Administration, USA; Jul. 1998; 30p; In English
Report No.(s): PB98-163298; DOT/FAA/AR-TN98/32; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report describes recent research by the Federal Aviation Administration (FAA) related to cargo compartment fire protection in large transport aircraft. A gaseous hydrofluorocarbon, HFC-125, was compared to Halon 1301 in terms of fire suppression effectiveness and agent decomposition levels in the cargo compartment and passenger cabin during full-scale tests involving a bulk-loaded cargo fire. Also, a zoned water mist system was designed and evaluated against a bulk-loaded cargo fire. An exploding aerosol can simulator is being developed to provide a repeatable fire threat for evaluation of new halon replacements agents. The potential severity of an exploding aerosol can inside a cargo compartment and the effectiveness of Halon 1301 inerting was demonstrated. Tests were also conducted to determine the effectiveness of Halon 1201 against a carbon fire involving oxygen canisters. Finally, HFC-125 was evaluated for use as a simulant for Halon 1301 during cargo compartment approval testing to demonstrate compliance with applicable FAA regulations.
NTIS
Commercial Aircraft; Fire Prevention; Transport Aircraft; Aerosols

19980221795 **Federal Aviation Administration, Fire Safety Section, Atlantic City, NJ USA**
Effects of Concentrated Hydrochloric Acid Spills and Aircraft Aluminum Skin
Speitel, L. C., Federal Aviation Administration, USA; Jul. 1998; 18p; In English
Report No.(s): PB98-163280; DOT/FAA/AR-TN97/108; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The purpose of this study is to evaluate the effect of a spill of concentrated hydrochloric acid (HCL) on the aircraft aluminum skin of a cargo compartment and to determine the time required for a spill to cause catastrophic failure for a worst-case scenario.
NTIS
Hydrochloric Acid; Aluminum; Cargo

19980223583 **NASA Langley Research Center, Hampton, VA USA**
Some Information of the Operational Experiences of Turbine-Powered Commercial Transports
Jewel, Joseph W., Jr., NASA Langley Research Center, USA; Hunter, Paul A., NASA Langley Research Center, USA; McLaughlin, Milton D., NASA Langley Research Center, USA; Jul. 20, 1961; 26p; In English
Report No.(s): NASA-TM-SX-595; L-1696; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report presents a brief discussion of some information on the operational experiences noted on VGH records from six types of turbine-powered commercial transport aircraft. These flight characteristics cover oscillatory motions, maneuver accelerations, sinking speeds, placard speed exceedances, and miscellaneous or unusual flight events.
Author
Commercial Aircraft; Flight Characteristics; Turbine Engines

19980223608 NASA Lewis Research Center, Cleveland, OH USA

Analytical Study of Soft Landings on Gas-Filled Bags

Espar, Jack B., NASA Lewis Research Center, USA; Morgan, William C., NASA Lewis Research Center, USA; Jan. 01, 1960; 32p; In English

Report No.(s): NASA-TR-R-75; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An analytical procedure was developed that is valid for bags of various arbitrary shapes and is applicable to planetary or lunar landings for sinking speeds that are small compared to the sonic velocity of the gas within the bag. For landing on the earth at speeds consistent with normal parachute descent, the relative merits of four bag shapes were evaluated both with and without gas bleed from the bags. Deceleration and onset rates acceptable for well-supported humans seem feasible.

Author (revised)

Soft Landing; Gas Bags; Parachute Descent; Descent Trajectories

19980223612 NASA Langley Research Center, Hampton, VA USA

Status of Spin Research for Recent Airplane Designs

Neihouse, Anshal I., NASA Langley Research Center, USA; Klinar, Walter J., NASA Langley Research Center, USA; Scher, Stanley H., NASA Langley Research Center, USA; 1960; 58p; In English

Report No.(s): NASA-TR-R-57; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

This report presents the status of spin research for recent airplane designs as interpreted at the Langley Research Center of the National Aeronautics and Space Administration. Major problem areas discussed include: (1) Interpretation of results of spin-model research (2) Analytical spin studies (3) Techniques involved in obtaining measurements of various parameters in the spin (4) Effectiveness of controls during spins and recoveries (5) Influence of long noses, strakes, and canards on spin and recovery characteristics (6) Correlation of spin and recovery characteristics for recent airplane and model designs. Analyses conclusions are drawn.

Author

Aircraft Spin; Spin Dynamics; Aerodynamic Characteristics; Aerodynamic Stalling; Aircraft Stability; Flight Characteristics

19980223919 NASA Dryden Flight Research Center, Edwards, CA USA

An Overview of an Experimental Demonstration Aerotow Program

Murray, James E., NASA Dryden Flight Research Center, USA; Bowers, Albion H., NASA Dryden Flight Research Center, USA; Lokos, William A., NASA Dryden Flight Research Center, USA; Peters, Todd L., NASA Dryden Flight Research Center, USA; Gera, Joseph, Analytical Services and Materials, Inc., USA; Sep. 1998; 29p; In English; 30th, 15-17 Sep. 1998, Reno, NV, USA; Sponsored by Society of Flight Test Engineers, USA

Contract(s)/Grant(s): RTOP 242-33-02-00-25

Report No.(s): NASA/TM-1998-206566; H-2279; NAS 1.15:206566; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An overview of an experimental demonstration of aerotowing a delta-wing airplane with low-aspect ratio and relatively high wing loading is presented. Aerotowing of future space launch configurations is a new concept, and the objective of the work described herein is to demonstrate the aerotow operation using an airplane configuration similar to conceptual space launch vehicles. Background information on the use of aerotow for a space launch vehicle is presented, and the aerotow system used in this demonstration is described. The ground tests, analytical studies, and flight planning used to predict system behavior and to enhance flight safety are detailed. The instrumentation suite and flight test maneuvers flown are discussed, preliminary performance is assessed, and flight test results are compared with the preflight predictions.

Author

Tetherlines; Towing; Tethering; Delta Wings; Flight Tests; Launch Vehicles; Low Aspect Ratio; Wing Loading

19980223930 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA

Flight Test Automation Options

Carico, Dean, Naval Air Warfare Center, USA; 1998; 13p; In English

Report No.(s): AD-A350677; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight testing is often called the key component of test and evaluation. The cost of conventional flight testing is expected to escalate approaching the 21st century and beyond. Augustine noted several years ago that if this trend continues, a single advanced fighter aircraft would cost more than the entire DoD budget by the middle of next century. As the cost of flight testing continues to escalate in a predicted hostile fiscal environment, it is important to consider options to help minimize flight test cost. Suggestions range from completely eliminating developmental testing to employing a variety of flight test automation options. Flight

test automation option concepts range from the fantasy of "push a button, the test is done," to the more practical use of a personal computer to help with some repetitive flight test tasks and to help store large amounts of related data. Options to help automate specific aspects of flight testing are starting to gain acceptance. Several test automation options exist that have the potential to enhance flight testing by permitting it to be done better, faster, cheaper, and safer. This paper briefly discusses a variety of flight test automation options including the OSD Automated Test Planning System (ATPS) work to automate the test and evaluation master plan (TEMP), the Army Test and Evaluation Planning and Reporting System (TEPRS), the G&C System work on Test
DTIC
Flight Tests; Data Acquisition; Test Ranges; Data Storage

19980223931 **Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA**
Telemetry and HPC Potential Applications
Normyle, Dennis, Naval Air Warfare Center, USA; 1998; 10p; In English
Report No.(s): AD-A350675; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

The intent of this paper is to familiarize the reader with the telemetry world and to investigate possible applications that the High Performance Computer (HPC) facility will provide for the T&E community. The following is a brief outline of the paper. 1) Current NAWC-AD Telemetry Capability - This section describes the current capability of RTPS (i.e. Number of PES rooms, Room Layout, Computers Used, Projects Supported, Number of flights flown etc....). 2) Anatomy of a F/A-18E1 Flutter flight - This section will describe the anatomy of a flutter flight, the type of maneuvers performed, the type of data collected, the AMOUNT of data collected, and how the data is processed post flight. 3) Telemetry/ACETEF Applications. A brief discussion on how RTPS and Manned Flight Simulator are currently linked and some of the early applications used. 4) Possible applications for HPC in Telemetry applications. 5) Conclusion.
DTIC
Telemetry; Utilization; Flight Tests

19980223932 **Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA**
Future Naval UCAV Applications & Enabling Technologies
Booz, Julieta E., Naval Air Warfare Center, USA; 1998; 12p; In English
Report No.(s): AD-A350673; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Briefing notes on the applications and enabling technologies for future naval Unmanned Combat Air Vehicles.
DTIC
Remotely Piloted Vehicles; Technologies

19980223943 **Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA**
Naval Rotary Wing Aircraft Flight Test Squadron Flight Test Approval Process
Mertaugh, Lawrence J., Naval Air Warfare Center, USA; Jan. 1998; 6p; In English
Report No.(s): AD-A350674; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

This presentation will provide a description of the process used by the Naval Rotary Wing Aircraft Test Squadron, at Patuxent River, for minimizing the risk associated with its flight test operations. This process is defined in terms of three basic functions. These functions are: Test Plan, Flight Clearance, and the Aircraft Modification/Configuration Control Sheet. It is through these functions that we provide oversight of the test planning, insure that any required aircraft modifications are sound, and provide controls over the aircraft modification process. Each of these functions play a role throughout the test program in preventing changes in testing that could jeopardize the quality of the test results or the safety of the crew or the aircraft.
DTIC
Rotary Wing Aircraft; Flight Tests; Risk

19980223965 **NASA Langley Research Center, Hampton, VA USA**
In-Flight System Identification
Morelli, Eugene A., NASA Langley Research Center, USA; 1998; 10p; In English; Atmospheric Flight Mechanics, 10-12 Aug. 1998, Boston, MA, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Report No.(s): AIAA Paper 98-4261; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

A method is proposed and studied whereby the system identification cycle consisting of experiment design and data analysis can be repeatedly implemented aboard a test aircraft in real time. This adaptive in-flight system identification scheme has many advantages, including increased flight test efficiency, adaptability to dynamic characteristics that are imperfectly known a priori, in-flight improvement of data quality through iterative input design, and immediate feedback of the quality of flight test results.

The technique uses equation error in the frequency domain with a recursive Fourier transform for the real time data analysis, and simple design methods employing square wave input forms to design the test inputs in flight. Simulation examples are used to demonstrate that the technique produces increasingly accurate model parameter estimates resulting from sequentially designed and implemented flight test maneuvers. The method has reasonable computational requirements, and could be implemented aboard an aircraft in real time.
Author
Design Analysis; Flight Tests; Dynamic Characteristics; Feedback

19980223968 **NASA Langley Research Center, Hampton, VA USA**
Subsonic Flight Tests of a 1/7-Scale Radio-Controlled Model of the North American X-15 Airplane with Particular Reference to High Angle-of-Attack Conditions
Hewes, Donald E., NASA Langley Research Center, USA; Hassell, James L., Jr., NASA Langley Research Center, USA; Jun. 1960; 46p; In English
Report No.(s): NASA-TM-X-283; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the subsonic stability and control characteristics of an unpowered 1/7-scale model based on the North American X-15 airplane was conducted by using a radio-controlled model launched from a helicopter and flown in free-gliding flight. At angles of attack below about 20 deg. where the model motions represent those of the X-15 airplane, the model was found to be both longitudinally and laterally stable, and the all-movable tail surfaces were found to be very effective. The model could also be flown at much higher angles of attack where the model motions did not necessarily represent those of the airplane because of slight geometrical differences and Reynolds number effects, but these test results are useful in evaluating the effectiveness at these angles of the type of lateral control system used in the X-15 airplane. In some cases, the model was flown to angles of attack as high as 60 or 70 deg. without encountering divergent or uncontrollable conditions. For some flights in which the model was subjected to rapid maneuvers, spinning motions were generated by application of corrective controls to oppose the direction of rotation. Rapid recoveries from this type of motion were achieved by applying roll control in the direction of rotation.
Author
Subsonic Speed; Flight Tests; Scale Models; Angle of Attack; Free Flight; Aerodynamic Stability; Aircraft Control; X-15 Aircraft

19980223972 **NASA Langley Research Center, Hampton, VA USA**
Aerodynamic Characteristics of a Target Drone Vehicle at Mach Numbers from 1.57 to 2.10
Blair, A. B., Jr., NASA Langley Research Center, USA; Fournier, Roger H., NASA Langley Research Center, USA; Jul. 1968; 72p; In English
Report No.(s): NASA-TM-SX-1531; AF-AM-627; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation of a 1/4-scale supersonic target drone model was performed in the Langley Unitary Plan wind tunnel to determine the effects of various sizes of canards, vertical tails, and ailerons on the aerodynamic characteristics. The tests were made at Mach numbers from 1.57 to 2.10 through an angle-of-attack range from about -5 to 23 deg.
Author
Aerodynamic Characteristics; Aircraft Structures; Control Surfaces; Supersonic Speed; Wind Tunnel Tests; Scale Models; Drone Vehicles; Wind Tunnel Models

19980223973 **NASA Langley Research Center, Hampton, VA USA**
Aerodynamic Characteristics of a Revised Target Drone Vehicle at Mach Numbers from 1.60 to 2.86
Blair, A. B., Jr., NASA Langley Research Center, USA; Babb, C. Donald, NASA Langley Research Center, USA; Feb. 1968; 54p; In English
Report No.(s): NASA-TM-SX-1532; L-5824; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley Unitary Plan wind tunnel to determine the aerodynamic characteristics of a revised target drone vehicle through a Mach number range from 1.60 to 2.86. The vehicle had canard surfaces and a swept clipped-delta wing with twin tip-mounted vertical tails.
Author
Aerodynamic Characteristics; Delta Wings; Canard Configurations; Drone Vehicles; Swept Wings; Wind Tunnel Tests; Supersonic Speed; Wind Tunnel Models; Aerodynamic Stability

19980223993 NASA Ames Research Center, Moffett Field, CA USA

Flight Investigation of the Low-Speed Characteristics of a 45 deg Swept-Wing Fighter-Type Airplane with Blowing Boundary-Layer Control Applied to the Leading- and Trailing-Edge Flaps

Quigley, Hervey C., NASA Ames Research Center, USA; Anderson, Seth B., NASA Ames Research Center, USA; Innis, Robert C., NASA Ames Research Center, USA; Sep. 1960; 46p; In English

Report No.(s): NASA-TN-D-321; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation has been conducted to study how pilots use the high lift available with blowing-type boundary-layer control applied to the leading- and trailing-edge flaps of a 45 deg. swept-wing airplane. The study includes documentation of the low-speed handling qualities as well as the pilots' evaluations of the landing-approach characteristics. All the pilots who flew the airplane considered it more comfortable to fly at low speeds than any other F-100 configuration they had flown. The major improvements noted were the reduced stall speed, the improved longitudinal stability at high lift, and the reduction in low-speed buffet. The study has shown the minimum comfortable landing-approach speeds are between 120.5 and 126.5 knots compared to 134 for the airplane with a slatted leading edge and the same trailing-edge flap. The limiting factors in the pilots' choices of landing-approach speeds were the limits of ability to control flight-path angle, lack of visibility, trim change with thrust, low static directional stability, and sluggish longitudinal control. Several of these factors were found to be associated with the high angles of attack, between 13 deg. and 15 deg., required for the low approach speeds. The angle of attack for maximum lift coefficient was 28 deg.

Author

Swept Wings; Fighter Aircraft; Boundary Layer Control; Externally Blown Flaps; Aerodynamic Coefficients; Aerodynamic Characteristics; Longitudinal Stability; Directional Stability

19980227097 NASA Flight Research Center, Edwards, CA USA

Flight Investigation of the Lift and Drag Characteristics of a Swept-Wing, Multijet, Transport-Type Airplane

Tambor, Ronald, NASA Flight Research Center, USA; Sep. 1960; 28p; In English

Report No.(s): NASA-TN-D-30; H-119; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The lift and drag characteristics of a Boeing KC-135 airplane were determined during maneuvering flight over the Mach number range from 0.70 to 0.85 for the airplane in the clean configuration at an altitude of 26,000 feet. Data were also obtained over the speed range of 130 knots to 160 knots at 9,000 feet for various flap deflections with gear down.

Author

C-135 Aircraft; Swept Wings; Lift; Aerodynamic Drag; Subsonic Speed; Aerodynamic Characteristics; Jet Aircraft; Flight Tests

19980227163 Army Command and General Staff Coll., Fort Leavenworth, KS USA

The Implications of Video Datalink on the AC-130, *5 Aug. 1997 - 5 Jun. 1998*

Hicks, John M., Army Command and General Staff Coll., USA; Jun. 05, 1998; 95p; In English

Report No.(s): AD-A350132; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This study considers the implications of video datalink (VDL) on the AC-130. Gunships use infrared and low-light television sensors, and synthetic aperture radar to search for and to identify target for close air support and interdiction missions. The addition of VDL offers gunship crews the ability to employ real-time information to the cockpit/offboard targeting (RTIC/OT) technology to improve situational awareness, survivability, and operational flexibility. Also, VDL offers the joint force air component commander (JFACC) inflight tasking capability, increased reconnaissance capability, operational flexibility and situation awareness. Ultimately, VDL allows command and control elements to exercise direct control of gunship operations. These capabilities are beneficial when they provide information to the crew or to the JFACC. However, VDL used to provide direct control of gunship operations may violate the Air Force doctrinal tenet of centralized control and decentralized execution. Lessons learned from recent contingencies, leadership doctrine, academic works on leadership and management theory all suggest that direct control of tactical mission can cause decreased survivability, ineffective span of control, task saturation, tactical inflexibility, mistrust between commanders and subordinates, decreased morale, and subordinates that lack initiative. The study provides recommendations to mitigate potential problems associated with the use of VDL on gunships.

DTIC

Data Links; Video Data; Television Systems; Alternating Current; Information Transfer; Infrared Detectors; Video Signals

19980227164 Army Command and General Staff Coll., Fort Leavenworth, KS USA

An Analysis of Prime Vendor Support for the AH64 Apache

Angelo, Anthony W., Army Command and General Staff Coll., USA; Jun. 05, 1998; 95p; In English

Report No.(s): AD-A350080; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

This study investigates the use of prime vendor support for the Army's AH64 Apache helicopter. It defines the term prime vendor support and it analyzes the reasons for applying this concept of logistical support to the Army's aviation support doctrine. This study shows why privatization of supply parts management has become not only necessary, but a driving force in the development and future application of prime vendor support. This study concludes that prime vendor support can create an innovative partnership between the Army and the Apache's prime vendor that will minimize the time it takes to deliver parts to mechanics and delay the purchasing of parts until they are needed to complete repairs. However, as the Army pursues a strategy of transformation needed to get from today's multiecheloned logistics system to more streamlined and efficient processes of support it must proceed with extreme caution. The complexities of resource management and the effects of changing existing processes to new concepts of support mandate further analysis and the development of procedures that will mitigate the risks associated with prime vendor support.
DTIC
Helicopters; AH-64 Helicopter; Investigation; Logistics Management

19980227171 **Army Command and General Staff Coll., Fort Leavenworth, KS USA**
Air Superiority Fighter Characteristics
Browne, James S., Army Command and General Staff Coll., USA; Jun. 05, 1998; 106p; In English
Report No.(s): AD-A350022; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

This study determines the essential characteristics of an air superiority fighter. Its importance stems from the assumption that air superiority is paramount in any military operation and that fighter aircraft play a major role. Air superiority as well as roles, functions, and missions are defined in chapter one to develop an understanding of the operative terms and definitions used throughout the thesis. This thesis is an in-depth study of the historical characteristics of the air superiority fighter. A complete review of air superiority fighter evolution is divided into four distinct generations. The review includes example aircraft that highlight the consistent characteristics found in each generation. The thesis research and analysis chapters focus on three key areas of interest. They are: (1) aircraft design, (2) avionics and weapons, and (3) training. The key areas of interest are coupled with a discussion of cost considerations during analysis. Fiscal constraints are a major factor in design and employment limitations. The thesis concludes that there are three essential characteristics of an air superiority fighter: (1) the aircraft is designed for the air-to-air role, (2) the aircraft has the first launch opportunity, and (3) the aircraft is flown by singularly trained air-to-air pilots.
DTIC
Cost Analysis; Aircraft Design; Fighter Aircraft

19980227179 **NASA Langley Research Center, Hampton, VA USA**
Summary of V-G and VGH Data Collected on Lockheed Electra Airplanes During Airplane Operations
Jewel, Joseph W., Jr., NASA Langley Research Center, USA; Fetner, Mary W., NASA Langley Research Center, USA; 1961; 60p; In English
Report No.(s): NASA-TM-SX-523; L-1467; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Data obtained by NASA VGH and V-G recorders on several Lockheed Electra airplanes operated over three domestic routes have been analyzed to determine the in-flight accelerations, airspeed practices, and landing accelerations experienced by this particular airplane. The results indicate that the accelerations caused by gusts and maneuvers are comparable to corresponding results for piston-engine transport airplanes. Oscillatory accelerations (apparently caused by the autopilot or control system) appear to occur about one-tenth as frequently as accelerations due to gusts. Airspeed operating practices in rough air generally follow the trends shown by piston-engine transports in that there is no significant difference between the average airspeed in rough or smooth air. Placard speeds were exceeded more frequently by the Electra airplane than by piston-engine transport airplanes. Generally, the landing-impact accelerations were higher than those for piston-engine transports.
Author
Transport Aircraft; Commercial Aircraft; Airline Operations; Impact Loads; Landing Loads

19980227196 **NASA Langley Research Center, Hampton, VA USA**
Summary of Flight-Test Results of the VZ-2 Tilt-Wing Aircraft
Pegg, Robert J., NASA Langley Research Center, USA; Feb. 1962; 44p; In English
Report No.(s): NASA-TN-D-989; L-1574; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight-test information gained from a tilt-wing research aircraft tested at the Langley Research Center has shown that design problems exist in such fields as low-speed stability and control, handling qualities, and flow separation during transition. The control power in the near-hovering configuration was considered by the pilots to be inadequate in yaw, marginal in pitch, and excessive in roll. Solutions for some of the design problems are indicated; for example, the addition of a leading-edge droop to the wing in an attempt to delay flow separation resulted in such significantly improved handling qualities in the transition range that an additional descent capability of 1,100 feet per minute was obtained.

Author

Flight Tests; Boundary Layer Separation; Vz-2 Aircraft; Separated Flow; Design Analysis; Leading Edges; Hovering

19980227205 NASA Ames Research Center, Moffett Field, CA USA

Aerodynamic Performance and Static Stability at Mach Number 3.3 of an Aircraft Configuration Employing Three Triangular Wing Panels and a Body Equal Length

James, Carlton S., NASA Ames Research Center, USA; Aug. 1960; 40p; In English

Report No.(s): NASA-TN-D-330; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An aircraft configuration, previously conceived as a means to achieve favorable aerodynamic stability characteristics., high lift-drag ratio, and low heating rates at high supersonic speeds., was modified in an attempt to increase further the lift-drag ratio without adversely affecting the other desirable characteristics. The original configuration consisted of three identical triangular wing panels symmetrically disposed about an ogive-cylinder body equal in length to the root chord of the panels. This configuration was modified by altering the angular disposition of the wing panels, by reducing the area of the panel forming the vertical fin, and by reshaping the body to produce interference lift. Six-component force and moment tests of the modified configuration at combined angles of attack and sideslip were made at a Mach number of 3.3 and a Reynolds number of 5.46 million. A maximum lift-drag ratio of 6.65 (excluding base drag) was measured at a lift coefficient of 0.100 and an angle of attack of 3.60. The lift-drag ratio remained greater than 3 up to lift coefficient of 0.35. Performance estimates, which predicted a maximum lift-drag ratio for the modified configuration 27 percent greater than that of the original configuration, agreed well with experiment. The modified configuration exhibited favorable static stability characteristics within the test range. Longitudinal and directional centers of pressure were slightly aft of the respective centroids of projected plan-form and side area.

Author

Aerodynamic Characteristics; Aircraft Configurations; Wing Panels; Aerodynamic Stability; Lift Drag Ratio; Delta Wings; Aerodynamic Drag; Aerodynamic Coefficients

19980227208 Naval Aerospace Medical Research Lab., Pensacola, FL USA

The Development and Initial Validation of the Unmanned Aerial Vehicle (UAV) External Pilot Selection System

Biggerstaff, S., Naval Aerospace Medical Research Lab., USA; Blower, D. J., Naval Aerospace Medical Research Lab., USA; Portman, C. A., Naval Aerospace Medical Research Lab., USA; Chapman, A. D., Naval Aerospace Medical Research Lab., USA; Mar. 05, 1998; 21p; In English

Report No.(s): AD-A350547; NAMRL-1398; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The purpose of this study was to develop physical and selection performance standards for the screening of candidates for entrance into the Unmanned Aerial Vehicle (UAV) Pioneer Pilot training program. A minimum Pioneer crew consists of an external pilot, internal pilot, and a mission commander/payload specialist. The mission commander/payload specialist is responsible for the overall planning and execution of the specific mission and control of the visual/information gathering during the mission. The internal pilot is responsible for the control of the Pioneer when it is beyond visual range. The external pilot is responsible for takeoffs, landings, and any in- visual-range control of the vehicle. A task analysis was done in the training and fleet squadrons to identify critical tasks for safe flight and the relevant skills required to perform the piloting tasks. From this task analysis, specific computer-based tests batteries were chosen as potential predictor variables. The system was programmed and students and external pilots were administered the test battery. A composite training measure was created from objective training scores, verified with subjective instructor ratings, and used as the criterion for predictive validation of the system. The sample size was small for the preliminary model, but a significant relationship between a composite of multitask tracking scores and UAV performance was observed (adjusted R2 = 0.86). In addition, structured and unstructured interviews of the Pioneer crews, students, instructors and senior squadron personnel were used to identity important physical characteristics essential for safe operation of the Pioneer. These traits were then used to derive medical screening criteria for all crew positions.

DTIC

Pilot Selection; Pilot Performance; Pilot Training; Proving; Pilotless Aircraft

19980227272 Naval Aerospace Medical Research Lab., Pensacola, FL USA

Landing Craft Air Cushion (LCAC) Navigator Selection System: Initial Model Development

Biggerstaff, S., Naval Aerospace Medical Research Lab., USA; Blower, D. J., Naval Aerospace Medical Research Lab., USA; Portman, C. A., Naval Aerospace Medical Research Lab., USA; Chapman, A., Naval Aerospace Medical Research Lab., USA; Mar. 05, 1998; 27p; In English

Report No.(s): AD-A350546; NAMRL-1399; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The LCAC is an amphibious hovercraft that can ride on a cushion of air across land or sea. Its control features are similar to a helicopter and it is designed to transport weapons, cargo, equipment and combat personnel. In the 1980s, the LCAC community was experiencing a high attrition rate, partially due to the absence of any valid selection mechanisms for crewmembers. The Naval Aerospace Medical Research Laboratory (NAMRL) developed a selection system for the LCAC operators and engineers and the system was transitioned to the Naval Operational Medicine Institute (NOMI) in 1992. Similar attrition problems were seen in the Navigator community and in FY 94-95 NAMRL was again tasked with developing a selection system. Concurrent validation of the system was done using 58 LCAC navigators. The preliminary predictive model was generated and the cut-off score derived from the sponsor's operational manpower needs. to date, 30 candidates have been screened with 25 being recommended for training. Thirteen of these candidates have entered training and completed the full 22 week syllabus. Five of the thirteen recommended candidates attrited during training, with multi-tasking as the main reason sited. The system is still being evaluated to: (1) possibly include more multi-tasking tests, (2) modify the predictive algorithm, and/or (3) raise the cut-off score to further reduce the attrition rates.

DTIC

Air Cushion Landing Systems; Navigators; Cushions; Ground Effect Machines

19980227280 **NASA Langley Research Center, Hampton, VA USA**

Effects of Boattailing and Nozzle Extension on the Thrust-Minus-Drag of a Multiple-Jet Configuration

Scott, William R., NASA Langley Research Center, USA; Jun. 1961; 50p; In English

Report No.(s): NASA-TN-D-887; L-862; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation of the effects of both boattailing and nozzle extension on the thrust-minus-drag of clustered-jet configurations has been conducted at Mach numbers from 0.60 to 1.40 and jet total-pressure ratios from 3 to 20. Three different boattails were tested: an 8 deg conical afterbody, a 16 deg circular-arc afterbody, and a third afterbody having a linear area variation with length. A cylindrical afterbody also was tested for comparison purposes. Extending from these bodies are four circular jet nozzles with a design Mach number of 2.5 which were spaced symmetrically about the body center line. The results indicated that an 8 deg conical afterbody provided the highest net thrust efficiency factors of the four models tested when the nozzle exits were at the optimum longitudinal location in each case. The other afterbodies in order of decreasing performance were the 16 deg circular-arc, the straight-line-area-distribution, and the cylindrical.

Author

Aerodynamic Drag; Thrust; Afterbodies; Boattails; Pressure Ratio; Cylindrical Bodies

19980227282 **NASA Dryden Flight Research Center, Edwards, CA USA**

Analysis of X-15 Landing Approach and Flare Characteristics Determined from the First 30 Flights

Matranga, Gene J., NASA Dryden Flight Research Center, USA; Jul. 1961; 54p; In English

Report No.(s): NASA-TN-D-1057; H-221; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The approach and flare maneuvers for the first 30 flights of the X-15 airplane and the various control problems encountered are discussed. The results afford a relatively good cross section of landing conditions that might be experienced with future glide vehicles having low lift-drag ratios. Flight-derived drag data show that preflight predictions based on wind-tunnel tests were, in general, somewhat higher than the values measured in flight. Depending on configuration, the peak lift-drag ratios from flight varied from 3.5 to 4.5 as compared with a predicted range of from 3.0 to 4.2. by employing [illegible] spiral-type patterns beginning at altitudes as high as 40,000 feet, the pilots were consistently able to touch down within about +/-1,000 feet of a designated point. A typical flare was initiated at a "comfortable" altitude of about 800 feet and an indicated airspeed of approximately 300 knots., which allowed a margin of excess speed. The flap and gear were extended when the flare was essentially completed, and an average touchdown was accomplished at a speed of about 185 knots indicated airspeed, an angle of attack of about 7 deg, and a rate of descent of about 4 feet per second. In general, the approach and landing characteristics were predicted with good accuracy in extensive preflight simulations. F-104 airplanes which simulated the X-15 landing characteristics were particularly valuable for pilot training.

Author

X-15 Aircraft; Aerodynamic Characteristics; Angle of Attack; Lift Drag Ratio; Landing; Touchdown; Descent; Approach

19980227307 **NASA Ames Research Center, Moffett Field, CA USA**

Experimental Investigation of a Hypersonic Glider Configuration at a Mach Number of 6 and at Full-Scale Reynolds Numbers

Seiff, Alvin, NASA Ames Research Center, USA; Wilkins, Max E., NASA Ames Research Center, USA; Jan. 1961; 74p; In English

Report No.(s): NASA-TN-D-341; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The aerodynamic characteristics of a hypersonic glider configuration, consisting of a slender ogive cylinder with three highly swept wings, spaced 120 apart, with the wing chord equal to the body length, were investigated experimentally at a Mach number of 6 and at Reynolds numbers from 6 to 16 million. The objectives were to evaluate the theoretical procedures which had been used to estimate the performance of the glider, and also to evaluate the characteristics of the glider itself. A principal question concerned the viscous drag at full-scale Reynolds number, there being a large difference between the total drags for laminar and turbulent boundary layers. It was found that the procedures which had been applied for estimating minimum drag, drag due to lift, lift curve slope, and center of pressure were generally accurate within 10 percent. An important exception was the non-linear contribution to the lift coefficient which had been represented by a Newtonian term. Experimentally, the lift curve was nearly linear within the angle-of-attack range up to 10 deg. This error affected the estimated lift-drag ratio. The minimum drag measurements indicated that substantial amounts of turbulent boundary layer were present on all models tested, over a range of surface roughness from 5 microinches maximum to 200 microinches maximum. In fact, the minimum drag coefficients were nearly independent of the surface smoothness and fell between the estimated values for turbulent and laminar boundary layers, but closer to the turbulent value. At the highest test Reynolds numbers and at large angles of attack, there was some indication that the skin friction of the rough models was being increased by the surface roughness. At full-scale Reynolds number, the maximum lift-drag ratio with a leading edge of practical diameter (from the standpoint of leading-edge heating) was 4.0. The configuration was statically and dynamically stable in pitch and yaw, and the center of pressure was less than 2-percent length ahead of the centroid of plan-form area.

Author

Hypersonic Gliders; Aerodynamic Characteristics; Aerodynamic Coefficients; Swept Wings; Hypersonic Speed; Laminar Boundary Layer; Turbulent Boundary Layer

19980227332 **NASA Langley Research Center, Hampton, VA USA**

An Evaluation of Effects of Flexibility on Wing Strains in Rough Air for a Large Swept-Wing Airplane by Means of Experimentally Determined Frequency-Response Functions with an Assessment of Random-Process Techniques Employed

Coleman, Thomas L., NASA Langley Research Center, USA; Press, Harry, NASA Langley Research Center, USA; Meadows, May T., NASA Langley Research Center, USA; 1960; 36p; In English

Report No.(s): NASA-TR-R-70; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight test measurements on a large swept-wing bomber airplane through rough air at altitudes of 5,000 and 35,000 feet are analyzed in order to determine the effects of airplane flexibility on wing bending and shear strains. For this purpose, the power spectra of the strain responses and the frequency-response functions for the strain responses to vertical gust disturbances are determined and compared with the strain responses for a quasi-rigid airplane. The measured power spectra and frequency-response functions are subject to distortion and statistical sampling errors from a variety of sources. A general analysis of the reliability of such results is presented and methods of estimating the distortions and sampling errors are developed. These methods are applied to the interpretation of the test results.

Author

Flight Tests; Errors; Sampling; Reliability; Distortion; Frequency Response

19980227344 **Federal Aviation Administration, Airworthiness Assurance Research and Development Branch, Atlantic City, NJ USA**

Vertical Drop Test of a Beechcraft 1900C Airliner *Final Report, Jul - Nov. 1995*

McGuire, Robert J.; Vu, Tong; May 1998; 83p; In English

Report No.(s): AD-A350509; AAR-431; DOT/FAA/AR-96/119; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

A commuter category Beechcraft 1900C airliner was subjected to a vertical impact drop test at the FAA William J. Hughes Technical Center, Atlantic City International Airport, New Jersey. The purpose of this test was to measure the impact response of the fuselage, cabin floor, cabin furnishings (including standard and modified seats), and anthropomorphic test dummies. The test was conducted to simulate the vertical velocity component of a severe but survivable crash impact. A low-wing, 19-passenger fuselage was dropped from a height of 11' 2" resulting in a vertical impact velocity of 26.8 ft/sec. The airframe was configured to simulate a typical flight condition, including seats (normal and experimental), simulated occupants, and cargo. For the test the wings were removed; the vertical and horizontal stabilizers were removed; the landing gear was removed; and the pilot and copilot seats were not installed. The data collected in the test and future tests will supplement the existing basis for improved seat and restraint systems for commuter category 14 Code of Federal Regulation (CFR) Part 23 airplanes. The test article was fully instrumented with accelerometers and load cells. Seventy-nine data channels were recorded. Results of the test are as follows: - the fuselage experienced an impact in the range of 149-160 g's, with an impact pulse duration of 9-10 milliseconds - the simulated occupants experienced g levels in the range of 32-45 g's with a pulse duration of 44-61 milliseconds - the test was considered to be a severe but definitely survivable impact - the fuselage structure maintained a habitable environment during and after the impact - the seat tracks remained attached to the fuselage along the entire length of the fuselage - all standard seats remained in their tracks after the impact - all exits remained operable

DTIC

Fuselages; Crashes; Damage Assessment; Impact Tests; Drop Tests; Airframes; Transport Aircraft; Wings; Aircraft Compartments

19980227362 **NASA Dryden Flight Research Center, Edwards, CA USA**

Measurements Obtained During the First Landing of the North American X-15 Research Airplane

McKay, James M., NASA Dryden Flight Research Center, USA; Oct. 1959; 38p; In English

Report No.(s): NASA-TM-X-207; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The first landing of the X-15 airplane was made at 8:43 a.m., June 8, 1959, on the hard surface of Rogers Dry Lake. One purpose of the first-glide flight was to evaluate the effectiveness of the landing-gear system. Some results are presented of the landing-approach characteristics, the impact period, and the runout phase of the landing maneuver. The results indicate that the touchdown was accomplished at a vertical velocity of 2.0 feet per second for the main gear and 13.5 feet per second for the nose gear. These vertical velocities were within the values of sinking speeds established by structural design limitations. However, permanent structural deformation occurred in the main-landing-gear system as a result of the landing, and a reevaluation of the gear is being made by the manufacturer. The landing occurred at a true ground speed of 158 knots for main-gear touchdown at an angle of attack of 8.50. The incremental acceleration at the main gear was 2.7g and 7.39 at the nose gear as a result of the landing. The incremental acceleration at the center of gravity of the airplane was 0.6g for the main-gear impact and 2.4g for the nose-gear impact. The incremental acceleration at the main gear as a result of the nose-gear impact was 4.8g. The extreme rearward location of the main-gear skids appears to offer satisfactory directional stability characteristics during the run- out phase of the landing. No evidence of nosewheel shimmy was indicated during the impact and runout phase of the landing despite the absence of a shimmy damper on the nose gear. The maximum amount of skid wear as a result of the landing was on the order of 0.005 inch. No appreciable amount of tire wear was indicated for the dual, corotating nosewheels.

Author

X-15 Aircraft; Structural Design; Directional Stability; Center of Gravity; Aerodynamic Characteristics; Angle of Attack; Deformation

19980227405 **NASA Langley Research Center, Hampton, VA USA**

Free-Flight Investigation of Radio Controlled Models with Parawings

Hewes, Donald E., NASA Langley Research Center, USA; Sep. 1961; 34p; In English

Report No.(s): NASA-TN-D-927; L-1374; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A free-flight investigation of two radio-controlled models with parawings, a glider configuration and an airplane (powered) configuration, was made to evaluate the performance, stability, and methods of controlling parawing vehicles. The flight tests showed that the models were stable and could be controlled either by shifting the center of gravity or by using conventional elevator and rudder control surfaces. Static wind-tunnel force-test data were also obtained.

Author

Parawings; Flight Tests; Free Flight; Wind Tunnel Tests; Aerodynamic Stability; Gliders; Aerodynamic Configurations; Aircraft Stability

19980227410 **NASA Langley Research Center, Hampton, VA USA**

Investigation of Low-Subsonic Flight Characteristics of a Model of a Hypersonic Boost-Glide Configuration Having a 78 deg. Delta Wing

Paulson, John W., NASA Langley Research Center, USA; Shanks, Robert E., NASA Langley Research Center, USA; May 1961; 32p; In English

Report No.(s): NASA-TN-D-894; L-452; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the low-subsonic stability and control characteristics of a model of a hypersonic boost-glide configuration having 78 deg. sweep of the leading edge has been made in the Langley full-scale tunnel. The model was flown over an angle-of-attack range from 10 to 35 deg. Static and dynamic force tests were made in the Langley free-flight tunnel. The investigation showed that the longitudinal stability and control characteristics were generally satisfactory with neutral or positive static longitudinal stability. The addition of artificial pitch damping resulted in satisfactory longitudinal characteristics being obtained with large amounts of static instability. The most rearward center-of-gravity position for which sustained flights could be made either with or without pitch damper corresponded to the calculated maneuver point. The lateral stability and control characteristics were satisfactory up to about 15 deg. angle of attack. The damping of the Dutch roll oscillation decreased with increasing angle of attack; the oscillation was about neutrally stable at 20 deg. angle of attack and unstable at angles of attack of about 25 deg. and above. Artificial damping in roll greatly improved the lateral characteristics and resulted in flights being made up to 35 deg. angle of attack.

Author

Flight Characteristics; Dynamic Stability; Boostglide Vehicles; Delta Wings; Swept Wings; Free Flight; Aircraft Control

19980227431 **NASA Langley Research Center, Hampton, VA USA**

Flight Tests of a 1/6-Scale Model of the Hawker P 1127 Jet VTOL Airplane

Smith, Charles C., Jr., NASA Langley Research Center, USA; 1961; 144p; In English

Report No.(s): NASA-TM-SX-531; L-1484; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

An experimental investigation has been made to determine the dynamic stability and control characteristics of a 1/6-scale flying model of the Hawker P IIP7 jet vertical-take-off-and-landing (VTOL) airplane in hovering and transition flight. The model was powered by a counter-rotating ducted fan driven by compressed-air jets at the tips of the fan blades. In hovering flight the model was controlled by jet-reaction controls which consisted of yaw and pitch jets at the extremities of the fuselage and a roll jet on each wing tip. In forward flight the model was controlled by conventional ailerons and rudder and an all-movable horizontal tail. In hovering flight the model could be flown smoothly and easily, but the roll control was considered too weak for rapid maneuvering or hovering in gusty air. Transitions from hovering to normal forward flight and back to hovering could be made smoothly and consistently and with only moderate changes in longitudinal trim. The model had a static longitudinal instability or pitch-up tendency throughout the transition range, but the rate of divergence in the pitch-up was moderate and the model could be controlled easily provided the angle of attack was not allowed to become too high. In both the transition and normal forward flight conditions the lateral motions of the model were difficult to control at high angles of attack, apparently because of low directional stability at small angles of sideslip. The longitudinal stability of the model in normal forward flight was generally satisfactory, but there was a decided pitch-up tendency for the flap-down condition at high angles of attack. In the VTOL landing approach condition, with the jets directed straight down or slightly forward, the nose-down pitch trim required was greater than in the transitions from hovering to forward flight, but the longitudinal instability was about the same. Take-offs and landings in still air could be made smoothly although there was a slight unfavorable ground effect on lift and a nose-down change in pitch trim near the ground. Short take-offs and landings could be made smoothly and consistently although the model experienced a decided nose-up change in pitching moment as it climbed out of ground effect.

Author

Flight Tests; Dynamic Stability; Vertical Takeoff Aircraft; Directional Stability; Scale Models; Hovering; Horizontal Flight; Aerodynamic Characteristics

19980227443 **Naval Postgraduate School, Monterey, CA USA**

Performance Enhancements to Joint Army/Navy Rotorcraft Analysis and Design (JANRAD) Software and Graphical User Interface (GUI)

Hucke, William L., Naval Postgraduate School, USA; Jun. 1998; 360p; In English

Report No.(s): AD-A350646; No Copyright; Avail: CASI; A16, Hardcopy; A03, Microfiche

The Joint Army/Navy Rotorcraft Analysis and Design (JANRAD) computer program was developed at the Naval Postgraduate School to perform performance, stability and control, and rotor dynamics analysis during preliminary helicopter design efforts. This thesis is the continuation of a previous work in which a Graphical User Interface (GUI) was developed and implemented as the front end of the NPS program. Due to the complexity of the GUI design, only the Performance module of JANRAD was completed by the prior student. This thesis expands the capabilities of the Performance module, and the JANRAD code, by adding graphical output of performance results, improved rotor sizing capabilities, resources for user defined blade elements and non-linear blade twist, airfoil meshing capabilities, and additional reference airfoil data corrected for compressibility effects. It contains the basic architecture for the Stability and Control module GUI. Additionally, utilizing actual UH-60A Black Hawk airfoil and test flight data as inputs, JANRAD version 5.0 was run to validate its output with the test flight results, and those produced in a

prior thesis by JANRAD version 3.1 (1995). Excellent agreement was demonstrated in all flight regimes. Utilizing airfoil data corrected for compressibility effects, high altitude runs resulted in much better correlation with test flight results than those experienced in 1995 using uncorrected airfoil data. A JANRAD Users Guide was updated and is included in Appendix A.
DTIC
Graphical User Interface; Computer Programs; Aircraft Design; Software Engineering; Helicopters

19980227452 **NASA Langley Research Center, Hampton, VA USA**
Incipient- and Developed-Spin and Recovery Characteristics of a Modern High-Speed Fighter Design with Low Aspect Ratio as Determined from Dynamic-Model Tests
Lee, Henry A., NASA Langley Research Center, USA; Libbey, Charles E., NASA Langley Research Center, USA; Dec. 1961; 22p; In English
Report No.(s): NASA-TN-D-956; L-1662; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Incipient- and developed-spin and recovery characteristics of a modern high-speed fighter design with low aspect ratio have been investigated by means of dynamic model tests. A 1/7-scale radio-controlled model was tested by means of drop tests from a helicopter. Several 1/25-scale models with various configuration changes were tested in the Langley 20-foot free-spinning tunnel. Model results indicated that generally it would be difficult to obtain a developed spin with a corresponding airplane and that either the airplane would recover of its own accord from any poststall motion or the poststall motion could be readily terminated by proper control technique. On occasion, however, the results indicated that if a post-stall motion were allowed to continue, a fully developed spin might be obtainable from which recovery could range from rapid to no recovery at all, even when optimum control technique was used. Satisfactory recoveries could be obtained with a proper-size tail parachute or strake, application of pitching-, rolling-, or yawing-moment rockets, or sufficient differential deflection of the horizontal tail.
Author
Fighter Aircraft; Low Aspect Ratio; Aircraft Design; Wind Tunnel Tests; Scale Models; Stability Derivatives; Dynamic Stability; Aerodynamic Stability; Drop Tests

19980227737 **NASA Langley Research Center, Hampton, VA USA**
Analysis of Effects of Interceptor Roll Performance and Maneuverability on Success of Collision-Course Attack
Phillips, William H., NASA Langley Research Center, USA; Aug. 1961; 42p; In English
Report No.(s): NASA-TN-D-952; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An attempt has been made to determine the importance of rolling performance and other factors in the design of an interceptor which uses collision-course tactics. A graphical method is presented for simple visualization of attack situations, by means of diagrams showing vectoring limits, that is, the ranges of inerceptor position and heading from which attacks may be successfully completed, the relative importance of rolling performance and normal-acceleration capability in determining the success of attacks is illustrated. The results indicate that the reduction in success of attacks due to reduced rolling performance (within the limits generally acceptable from the pilots' standpoint) is very small, whereas the benefits due to substantially increasing the normal-acceleration capability are large. Additional brief analyses show that the optimum speed for initiating a head-on attack is often that corresponding to the upper left-hand corner of the V-g diagram. In these cases, increasing speed beyond this point for given values of normal acceleration and radar range rapidly decreases the width of the region from which successful attacks can be initiated. On the other hand, if the radar range is increased with a variation somewhere between the first and second power of the interceptor speed, the linear dimensions of the region from which successful attacks can be initiated vary as the square of the interceptor speed.
Author
Roll; Lateral Control; Maneuverability; Interceptors; Fighter Aircraft; Collisions; Aircraft Performance

19980227749 **NASA Langley Research Center, Hampton, VA USA**
Some Landing Studies Pertinent to Glider-Reentry Vehicles
Houbolt, John C., NASA Langley Research Center, USA; Batterson, Sidney A., NASA Langley Research Center, USA; Aug. 1960; 24p; In English
Report No.(s): NASA-TN-D-448; L-1066; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results are presented of some landing studies that may serve as guidelines in the consideration of landing problems of glider-reentry configurations. The effect of the initial conditions of sinking velocity, angle of attack, and pitch rate on impact severity and the effect of locating the rear gear in various positions are discussed. Some information is included regarding the influence

of landing-gear location on effective masses. Preliminary experimental results on the slideout phase of landing include sliding and rolling friction coefficients that have been determined from tests of various skids and all-metal wheels.
Author
Gliders; Reentry Vehicles; Spacecraft Landing; Spacecraft Reentry

19980227754 NASA Langley Research Center, Hampton, VA USA
Effect of Blade Cutout on Power Required by Helicopters Operating at High Tip-Speed Ratios
Gessow, Alfred, NASA Langley Research Center, USA; Gustafson, F. B., NASA Langley Research Center, USA; Sep. 1960; 20p; In English
Report No.(s): NASA-TN-D-382; L-696; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A numerical study was made of the effects of blade cutout on the power required by a sample helicopter rotor traveling at tip-speed ratios of 0.3, 0.4, and 0.5. The amount of cutout varied from 0 to 0.5 of the rotor radius and the calculations were carried out for a thrust coefficient-solidity ratio of 0.04. In these calculations the blade within the cutout radius was assumed to have zero chord. The effect of such cutout on profile-drag power ranged from almost no effect at a tip-speed ratio of 0.3 to as much as a 60 percent reduction at a tip-speed ratio of 0.5. Optimum cutout was about 0.3 of the rotor radius. Part of the large power reduction at a tip-speed ratio of 0.5 resulted from a reduction in tip-region stall, brought about by cutout. For tip-speed ratios greater than 0.3, cutout also effected a significant increase in the ability of the rotor to overcome helicopter parasite drag. It is thus seen that the adverse trends (at high tip-speed ratios) indicated by the uniform-chord theoretical charts are caused in large measure by the center portion of the rotor. The extent to which a modified-design rotor can actually be made more efficient at high speeds than a uniform-chord rotor will depend in practice on the degree of success in minimizing the blade plan form near the center and on special modifications in center-section profiles. A few suggestions and estimates in regard to such modifications are included herein.
Author
Helicopters; High Speed; Tip Speed; Rotor Blades (Turbomachinery); Thrust; Openings

19980227756 NASA Langley Research Center, Hampton, VA USA
A Flight Study of the Conversion Maneuver of a Tilt-Duct VTOL Aircraft
Tapscott, Robert J., NASA Langley Research Center, USA; Kelley, Henry L., NASA Langley Research Center, USA; Nov. 1960; 14p; In English
Report No.(s): NASA-TN-D-372; L-891; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight records are presented from an early flight test of a wing-tip mounted tilting-ducted-fan, vertical take-off and landing (VTOL) aircraft configuration. Time histories of the aircraft motions, control positions, and duct pitching-moment variation are presented to illustrate the characteristics of the aircraft in hovering, in conversion from hovering to forward flight, and in conversion from forward flight to hovering. The results indicate that during essentially continuous slow level- flight conversions, this aircraft experiences excessive longitudinal trim changes. Studies have shown that the large trim changes are caused primarily by the variation of aerodynamic moments acting on the duct units. Action of the duct-induced downwash on the horizontal stabilizer during the conversion also contributes to the longitudinal trim variations. Time histories of hovering and slow vertical descent in the final stages of landing in calm air show angular motions of the aircraft as great as +/- 10 deg. about all axes. Stick and pedal displacements required to control the aircraft during the landing maneuver were on the order of 50 to 60 percent of the total travel available.
Author
Vertical Takeoff Aircraft; Aircraft Configurations; Aircraft Control; Flight Tests; Ducted Fans; Flight Characteristics

19980227770 NASA Langley Research Center, Hampton, VA USA
Preliminary Investigation of a Paraglider
Rogallo, Francis M., NASA Langley Research Center, USA; Lowry, John G., NASA Langley Research Center, USA; Croom, Delwin R., NASA Langley Research Center, USA; Taylor, Robert T., NASA Langley Research Center, USA; Aug. 1960; 28p; In English
Report No.(s): NASA-TN-D-443; L-827; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A preliminary investigation of the aerodynamic and control characteristics of a flexible glider similar to a parachute in construction has been made at the Langley Research Center to evaluate its capabilities as a reentry glider. Preliminary weight estimates of the proposed vehicle indicate that such a structure can be made with extremely low wing loading. Maximum temperatures during the reentry maneuver might be held as low as about 1,500 F. The results of wind-tunnel and free-glide tests show that the glider when constructed of nonporous material performed extremely well at subsonic speeds and could be flown at angles of attack

from about 200 to 900. At supersonic speeds the wing showed none of the unfavorable tendencies exhibited by conventional parachutes at these speeds, such as squidding and breathing. Several methods of packing and deploying the glider have been successfully demonstrated. The results of this study indicate that this flexible-lifting-surface concept may provide a lightweight controllable paraglider for manned space vehicles.
Author
Lifting Reentry Vehicles; Wind Tunnel Tests; Reentry; Parachutes; Paragliders; Supersonic Speed; Manned Spacecraft

19980227777 **NASA Langley Research Center, Hampton, VA USA**
Data from a Static-Thrust Investigation of Large-Scale General Research VTOL-STOL Model in Ground Effect
Huston, Robert J., NASA Langley Research Center, USA; Winston, Matthew M., NASA Langley Research Center, USA; Aug. 1960; 66p; In English
Report No.(s): NASA-TN-D-397; L-987; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The model was tested at two different elevations with the wing pivot at 1.008 and 2.425 propeller diameters above the ground. The slipstream of the propellers was deflected by tilting the wing and propellers, by deflections of large-chord trailing-edge flaps, and by combinations of flap deflection and wing tilt. Tests were conducted over a range of propeller disk loadings from 7.41 to 29.70 pounds per square foot. Force data for the complete model and pressure distributions for the wing and flaps behind one propeller were recorded and are presented in tabular form without analysis.
Author
Vertical Takeoff Aircraft; Aircraft Configurations; Body-Wing Configurations; Aircraft Design; Short Takeoff Aircraft; Aircraft Structures; Flaps (Control Surfaces); Rotor Aerodynamics

19980227804 **NASA Langley Research Center, Hampton, VA USA**
Longitudinal Aerodynamic Characteristics of a Four-Propeller Deflected Slipstream VTOL Model Including the Effects of Ground Proximity
Kuhn, Richard E., NASA Langley Research Center, USA; Grunwald, Kalman J., NASA Langley Research Center, USA; Nov. 1960; 136p; In English
Report No.(s): NASA-TN-D-248; L-735; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

Results are presented of a wind-tunnel investigation of the longitudinal stability, control, and performance characteristics of a model of a four-propeller deflected-slipstream VTOL airplane in the transition speed range. These results indicate that steady level-flight transition and descending flight-path angles up to 7 or 8 deg. out of the region of ground effect can be accomplished without wing stall being encountered. In general, the pitching moments out of ground proximity can be adequately trimmed by programming the stabilizer incidence to increase with increasing flap deflection, except for a relatively large diving moment in the hovering condition. The deflection of the slipstream onto the horizontal tail in proximity of the ground substantially increases the diving moment in hovering, unless the tail is set at a large nosedown incidence.
Author
Aerodynamic Characteristics; Longitudinal Stability; Wind Tunnel Tests; Longitudinal Control; Vertical Takeoff Aircraft; Propeller Slipstreams; Flight Paths; Flapping

06
AIRCRAFT INSTRUMENTATION

Includes cockpit and cabin display devices; and flight instruments.

19980227268 **Naval Postgraduate Sc[illegible]erey, CA USA**
Avionics System Development for a [illegible]ing Unmanned Aerial Vehicle
Greer, Daniel S., Naval Postgraduate School, USA; Jun. 1998; 122p; In English
Report No.(s): AD-A350437; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

The Naval Postgraduate School has developed a successful Rapid Flight Test Prototyping System (RFTPS) for the development of software for remote computer control of fixed wing Unmanned Aerial Vehicles (UAV). This thesis reviews the work accomplished to mount sensors on a small remote controlled helicopter with instrumentation compatible with the RFTPS: an inertial measurement unit, a Global Positioning System (GPS) receiver, an altitude sensor and associated power supply and telemetry equipment. A helicopter with sufficient lift capability was selected and a lightweight aluminum structure was built to serve as both an avionics platform for the necessary equipment and also as a landing skid. Since the altitude sensors used for fixed wing UAV's, such as barometric sensors and GPS, do not provide sufficient accuracy for low altitude hover control, a lightweight, precision

altimeter was developed using ultrasound technology. Circuitry was developed to drive a Polaroid 6500 Series Ranging Module and process the output data in a form compatible with the RFTPS avionics architecture. Flight testing revealed severe vibrations throughout the helicopter. An alternative avionics package of reduced size was constructed to house the sonic altimeter and a three-axis accelerometer. Subsequent test flight results and recommendations for further research are provided.
DTIC
Avionics; Rotary Wings; Fixed Wings; Flight Tests

19980227412 NASA Langley Research Center, Hampton, VA USA
A Simulator Study of the Effectiveness of a Pilot's Indicator which Combined Angle of Attack and Rate of Change of Total Pressure as Applied to the Take-Off Rotation and Climbout of a Supersonic Transport
Hall, Albert W., NASA Langley Research Center, USA; Harris, Jack E., NASA Langley Research Center, USA; Sep. 1961; 24p; In English
Report No.(s): NASA-TN-D-948; L-1644; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A simulator study has been made to determine the effectiveness of a single instrument presentation as an aid to the pilot in controlling both rotation and climbout path in take-off. The instrument was basically an angle-of-attack indicator, biased with a total-pressure-rate input as a means of suppressing the phugoid oscillation. Linearized six-degree-of-freedom equations of motion were utilized in simulating a hypothetical supersonic transport as the test vehicle. Each of several experienced pilots performed a number of simulated take-offs, using conventional flight instruments and either an angle-of-attack instrument or the combined angle-of-attack and total-pressure-rate instrument. The pilots were able to rotate the airplane, with satisfactory precision, to the 15 deg. angle of attack required for lift-off when using either an angle-of-attack instrument or the instrument which combined total-pressure-rate with angle of attack. At least 4 to 6 second-S appeared to be required for rotation to prevent overshoot, particularly with the latter instrument. The flight paths resulting from take-offs with simulated engine failures were relatively smooth and repeatable within a reasonably narrow band when the combined angle-of-attack and total-pressure-rate instrument presentation was used. Some of the flight paths resulting from take-offs with the same engine-failure conditions were very oscillatory when conventional instruments and an angle-of-attack instrument were used. The pilots considered the combined angle-of-attack and total- pressure-rate instrument a very effective aid. Even though they could, with sufficient practice, perform satisfactory climbouts after simulated engine failure by monitoring the conventional instruments and making correction based on their readings, it was much easier to maintain a smooth flight path with the single combined angle-of-attack and total-pressure-rate instrument.
Author
Indicating Instruments; Flight Instruments; Angle of Attack; Takeoff; Simulators; Aircraft Pilots; Supersonic Transports; Pressure Distribution

19980227734 NASA Langley Research Center, Hampton, VA USA
Repeatability, Drift, and Aftereffect of Three Types of Aircraft Altimeters
Gracey, William, NASA Langley Research Center, USA; Stell, Richard E., NASA Langley Research Center, USA; Jul. 1961; 42p; In English
Report No.(s): NASA-TN-D-922; L-1580; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

In a series of laboratory tests of a number of sensitive altimeters 5 (Air Force type C-12 and C-13) and of precision altimeters (Air Force 8 type MA-1), the repeatability was determined for the full range of each type of instrument the drift characteristics were determined during 1-hour periods at various altitudes, and the drift and aftereffect were measured for a variety of simulated flights representative of some civil and military operations. For comparable altitude ranges, the repeatability errors of the C-12 and C-13 types were generally of the same order while those of the MA-1 type were somewhat smaller. The drift and aftereffect of the C-12 instruments were smaller than those of the C-13 instruments, and the drift and aftereffect of the MA-1 altimeters were considerably smaller than those of both types of the sensitive instruments. The drift of each of the three types of altimeters was found to increase with altitude and the drift of the precision type was found to increase with increasing rate of altitude change preceding the drift test.
Author
Altimeters; Drift (Instrumentation); Aircraft Instruments; Altimetry

19980227790 NASA Langley Research Center, Hampton, VA USA
Repeatability of the Over-All Errors of an Airplane Altimeter Installation in Landing-Approach Operations
Gracey, William, NASA Langley Research Center, USA; Stickle, Joseph W., NASA Langley Research Center, USA; May 1961; 22p; In English
Report No.(s): NASA-TN-D-898; L-1333; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight tests have been conducted to determine the repeatability of the over-all altimetry errors in the landing-approach condition of two sensitive altimeters (Air Force type C-12) installed in the cockpit of a transport airplane and of four precision altimeters (Air Force type MA-1) installed in a photo-observer. Data were obtained through a speed range of 62 to 100 knots during 42 landing-approach operations conducted on four different days. The results of the tests show that the repeatability errors of the two sensitive altimeters are +/- 35 feet and +/- 39 feet. These errors are of the same order as the maximum repeatability error measured in previous tests of eleven airplanes of the same type. For each of the four flights of the present tests the mean values of the data obtained with the two sensitive altimeters shifted by relatively large amounts, apparently because of the interaction of the stability and aftereffect- recovery characteristics of the instruments. For concurrent measurements of the over-all errors of the four precision altimeters, it is concluded that for comparable installations, the repeatability errors measured with these altimeters would be smaller than those measured with the sensitive altimeters.

Author

Altimeters; Transport Aircraft; Altimetry; Flight Tests; Aircraft Instruments; Landing Instruments; Installing

07
AIRCRAFT PROPULSION AND POWER

Includes prime propulsion systems and systems components, e.g., gas turbine engines and compressors; and onboard auxiliary power plants for aircraft.

19980221787 Naval Air Warfare Center, Weapons Div., China Lake, CA USA

FAA T53-L-13L Turbine Fragment Containment Test *Final Report*

Frankenberger, C. E., III, Naval Air Warfare Center, USA; Jun. 1998; 24p; In English

Contract(s)/Grant(s): DTFA03-95-X-90019

Report No.(s): PB98-159965; DOT/FAA/AR-98/22; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The result of the FAA T53-L-13L engine turbine disk fragment containment test is presented in this report. A containment ring was fabricated with a 0.014 inch titanium inner and outer sleeve. One-inch-thick Kevlar 29 ballistic fabric made up the primary structure of the containment ring. The ring was reinforced with titanium rods inserted through the fabric and laser welded to the inner and outer sleeves. The engine and containment ring were installed in an UH-1 Hey helicopter. The second stage power turbine disk was notched so that the disk would rupture at approximately 20,400 rpm. The engine was started and immediately accelerated to minimize the chance of a premature rupture. The event was recorded on high-speed film at 4000 pictures per second. The disk ruptured as the engine accelerated through 19,629 rpm. The disk ruptured into three equal section (approximately 3.6 lbs. each). The result was a contained tri-hub burst with minor bulging of the containment ring and little sign of distress to the airframe. This test demonstrated the capability to contain a tri-hub burst on a medium sized turboshaft helicopter engine.

NTIS

Turbines; Fragments; Containment; Turbine Engines

19980223961 NASA Dryden Flight Research Center, Edwards, CA USA

Flight Testing the Linear Aerospike SR-71 Experiment (LASRE)

Corda, Stephen, NASA Dryden Flight Research Center, USA; Neal, Bradford A., NASA Dryden Flight Research Center, USA; Moes, Timothy R., NASA Dryden Flight Research Center, USA; Cox, Timothy H., NASA Dryden Flight Research Center, USA; Monaghan, Richard C., NASA Dryden Flight Research Center, USA; Voelker, Leonard S., NASA Dryden Flight Research Center, USA; Corpening, Griffin P., NASA Dryden Flight Research Center, USA; Larson, Richard R., NASA Dryden Flight Research Center, USA; Powers, Bruce G., Analytical Services and Materials, Inc., USA; Sep. 1998; 24p; In English; 30th, 15-17 Sep. 1998, Reno, NV, USA; Sponsored by Society of Flight Test Engineers, USA

Contract(s)/Grant(s): RTOP 242-33-02-00-23

Report No.(s): NASA/TM-1998-206567; H-2280; NAS 1.15:206567; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The design of the next generation of space access vehicles has led to a unique flight test that blends the space and flight research worlds. The new space vehicle designs, such as the X-33 vehicle and Reusable Launch Vehicle (RLV), are powered by linear aerospike rocket engines. Conceived of in the 1960's, these aerospike engines have yet to be flown, and many questions remain regarding aerospike engine performance and efficiency in flight. to provide some of these data before flying on the X-33 vehicle and the RLV, a spacecraft rocket engine has been flight-tested atop the NASA SR-71 aircraft as the Linear Aerospike SR-71 Experiment (LASRE). A 20 percent-scale, semispan model of the X-33 vehicle, the aerospike engine, and all the required fuel and oxidizer tanks and propellant feed systems have been mounted atop the SR-71 airplane for this experiment. A major technical objective of the LASRE flight test is to obtain installed-engine performance flight data for comparison to wind-tunnel results and for the development of computational fluid dynamics-based design methodologies. The ultimate goal of firing the aerospike rocket engine in flight is still forthcoming. An extensive design and development phase of the experiment hardware has been completed, including approximately 40 ground tests. Five flights of the LASRE and firing the rocket engine using inert liquid nitrogen and helium in place of liquid oxygen and hydrogen have been successfully completed.

Author

SR-71 Aircraft; X-33 Reusable Launch Vehicle; Reusable Launch Vehicles; Flight Tests; Computational Fluid Dynamics; Aerospike Engines; Aerodynamic Characteristics

19980223990 **NASA Lewis Research Center, Cleveland, OH USA**

Effects of Tip Clearance and Casing Recess on Heat Transfer and Stage Efficiency in Axial Turbines

Ameri, A. A., AYT Corp., USA; Steinthorsson, E., NASA Lewis Research Center, USA; Rigby, David L., NYMA, Inc., USA; Aug. 1998; 15p; In English; Turbo, 2-5 Jun. 1998, Stockholm, Sweden; Sponsored by American Society of Mechanical Engineers, USA

Contract(s)/Grant(s): NAS3-27571; RTOP 523-26-13-00

Report No.(s): NASA/CR-1998-208514; E-11287; NAS 1.26:208514; ICOMP-98-04; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Calculations were performed to assess the effect of the tip leakage flow on the rate of heat transfer to blade, blade tip and casing. The effect on exit angle and efficiency was also examined. Passage geometries with and without casing recess were considered. The geometry and the flow conditions of the GE-E 3 first stage turbine, which represents a modern gas turbine blade were used for the analysis. Clearance heights of 0%, 1%, 1.5% and 3% of the passage height were considered. For the two largest clearance heights considered, different recess depths were studied. There was an increase in the thermal load on all the heat transfer surfaces considered due to enlargement of the clearance gap. Introduction of recessed casing resulted in a drop in the rate of heat transfer on the pressure side but the picture on the suction side was found to be more complex for the smaller tip clearance height considered. For the larger tip clearance height the effect of casing recess was an orderly reduction in the suction side heat transfer as the casing recess height was increased. There was a marked reduction of heat load and peak values on the blade tip upon introduction of casing recess, however only a small reduction was observed on the casing itself. It was reconfirmed that there is a linear relationship between the efficiency and the tip gap height. It was also observed that the recess casing has a small effect on the efficiency but can have a moderating effect on the flow underturning at smaller tip clearances.

Author

Gas Turbines; Heat Transfer; Axial Flow Turbines; Blade Tips; Clearances; Recesses

19980227187 **NASA Lewis Research Center, Cleveland, OH USA**

Investigation of the Effects of Low Reynolds Number Operation on the Performance of a Single-Stage Turbine with a Downstream Stator

Forrette, Robert E., NASA Lewis Research Center, USA; Holeski, Donald E., NASA Lewis Research Center, USA; Plohr, Henry W., NASA Lewis Research Center, USA; Sep. 1959; 52p; In English

Report No.(s): NASA-TM-X-9; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

High-altitude turbojet performance is adversely affected by the effects of low air density. This performance loss is evaluated as a Reynolds number effect, which represents the increased significance of high fluid viscous forces in relation to dynamic fluid forces as the Reynolds number is decreased. An analytical and experimental investigation of the effects of low Reynolds number operation on a single-stage, high-work-output turbine with a downstream stator was carried out at Reynolds numbers of 182,500, 39,600, and 23,000, based on average rotor-design flow conditions. At low Reynolds numbers and turbulent flow conditions, increased viscous losses caused decreased effective flow area, and thus decreased weight flow, torque, and over-all efficiency at a given equivalent speed and pressure ratio. Decreasing the Reynolds number from 182,500 to 23,000 at design equivalent speed resulted in a 5.00-point loss in peak over-all turbine efficiency for both theory and experiment. The choking equivalent weight flow decreased 2.30 percent for these conditions. Limiting loading work output was reached at design equivalent speed for all three Reynolds numbers. The value of limiting loading work output at design speed decreased 4.00 percent as Reynolds number was decreased from 182,500 to 23,000. A theoretical performance-prediction method using basic boundary-layer relations gave good agreement with experimental results over most of the performance range at a given Reynolds number if the experimental and analytical design operating conditions were carefully matched at the highest Reynolds number with regard to design perfor-

mance parameters. High viscous losses in the inlet stator and rotor prevented the attainment of design equivalent work output at the lowest Reynolds number of 23,000.
Author
Low Reynolds Number; Performance Prediction; Turbulent Flow; Turbojet Engines; Turbines; Design Analysis; Engine Design; Engine Parts

19980227324 Naval Air Warfare Center, Weapons Div., China Lake, CA USA
FAA T53-L-13L Turbine Fragment Containment Test *Final Report*
Frankenberger, C. E.; Jun. 1998; 16p; In English
Contract(s)/Grant(s): DTFA03-95-X-90019
Report No.(s): AD-A350454; DOT/FAA/AR-98/22; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The result of the FAA T53-L-13L engine turbine disk fragment containment test is presented in this report. A containment ring was designed and fabricated by Pepin Associates, Inc. and provided to the Naval Air Warfare Center, Weapons Division by the William J. Hughes Technical Center. This ring was fabricated with a 0.014-inch titanium inner and outer sleeve. One-inch-thick Kevlar 29 ballistic fabric made up the primary structure of the containment ring. The ring was reinforced with titanium rods inserted through the fabric and laser welded to the inner and outer sleeves. The engine and containment ring were installed in an UH-1 Huey helicopter. The second stage power turbine disk was notched so that the disk would rupture at approximately 20,400 rpm. The engine was started and immediately accelerated to minimize the chance of a premature rupture. The event was recorded on high-speed film at 4000 pictures per second. The disk ruptured as the engine accelerated through 19,629 rpm. The disk ruptured into three equal sections (approximately 3.6 lbs. each). The result was a contained tri-hub burst with minor bulging of the containment ring and little sign of distress to the airframe. This test demonstrated the capability to contain a tri-hub burst on a medium sized turboshaft helicopter engine.
DTIC
Helicopter Engines; Turbines; Rings; Fragments; Containment; Turboshafts

08
AIRCRAFT STABILITY AND CONTROL

Includes aircraft handling qualities; piloting; flight controls; and autopilots.

19980223075 NASA Langley Research Center, Hampton, VA USA
Exploratory Investigation at Mach Number of 2.01 of the Longitudinal Stability and Control Characteristics of a Winged Reentry Configuration
Foster, Gerald V., NASA Langley Research Center, USA; Dec. 1959; 22p; In English
Report No.(s): NASA-TM-X-178; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted to determine the longitudinal stability and control characteristics of a reentry configuration at a Mach number of 2.01. The configuration consisted of clipped delta wing with hinged wing-tip panels. The results indicate that deflecting the wing-tip panels from a position normal to the wing chord plane to a position coincident with the wing chord plane resulted in a stabilizing change in the pitching-moment characteristics but did not significantly affect the nonlinearity of the pitching-moment variation with angle of attack. The trailing-edge controls were effective in producing pitching moment throughout the angle-of-attack range for control deflections up to at least 600. The control deflection required for trim, however, varied nonlinearly with angle of attack. It would appear that this nonlinearity as well as the maximum deflection required for trim could be greatly decreased by utilizing a leading-edge control in conjunction with a trailing-edge control.
Author
Reentry Vehicles; Delta Wings; Longitudinal Stability; Supersonic Speed; Aerodynamic Stability; Spacecraft Stability; Wind Tunnel Stability Tests; Spacecraft Control

19980223588 NASA Langley Research Center, Hampton, VA USA
Transonic Flutter Investigation of Models of T-Tail of Blackburn NA-39 Airplane
Jones, George W., Jr., NASA Langley Research Center, USA; Farmer, Moses G., NASA Langley Research Center, USA; 1959; 54p; In English
Report No.(s): NASA-TM-SX-242; L-648; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A transonic flutter investigation has been made of models of the T-tail of the Blackburn NA-39 airplane. The models were dynamically and elastically scaled from measured airplane data in accordance with criteria which include a flutter safety margin.

The investigation was made in the Langley transonic blowdown tunnel and covered a Mach number range from 0.73 to 1.09 at simulated altitudes extending to below sea level. The results of the investigation indicated that, if differences between the measured model and scaled airplane properties are disregarded, the airplane with the normal value of stabilizer pitching stiffness should have a stiffness margin of safety of at least 32 percent at all Mach numbers and altitudes within the flight boundary. However, the airplane with the emergency value of stabilizer pitching stiffness would not have the required margin of safety from symmetrical flutter at Mach numbers greater than about 0.85 at low altitudes. First-order corrections for some differences between the measured model and scaled airplane properties indicated that the airplane with the normal value of stabilizer pitching stiffness would still have an adequate margin of safety from flutter and that the flutter safety margin for the airplane with the emergency value of stabilizer pitching stiffness would be changed from inadequate to adequate. However, the validity of the corrections is questionable.

Author

Transonic Flutter; Wind Tunnel Tests; Wind Tunnel Models; Tail Assemblies; Flutter Analysis; Aeroelasticity

19980223607 **NASA Langley Research Center, Hampton, VA USA**

The Lateral Response of Airplanes to Random Atmospheric Turbulence

Eggleston, John M., NASA Langley Research Center, USA; Phillips, William H., NASA Langley Research Center, USA; 1960; 62p; In English

Report No.(s): NASA-TR--R-74; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Random variations of gust velocities across the span and along the fuselage are considered. In part 1 a simplified method is presented in which the gust velocities are represented as rolling gusts, yawing gusts, and side gusts. A sample calculation procedure is presented for obtaining the response of the airplane in each degree of freedom.

Author

Atmospheric Turbulence; Gusts; Prediction Analysis Techniques; Yaw

19980223619 **NASA Ames Research Center, Moffett Field, CA USA**

An Examination of Handling Qualities Criteria for V/STOL Aircraft

Anderson, Seth B., NASA Ames Research Center, USA; Jul. 1960; 56p; In English

Report No.(s): NASA-TN-D-331; A-406; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A study has been undertaken to define hand-ling qualities criteria for V/STOL aircraft. With the current military requirements for helicopters and airplanes as a framework, modifications and additions were made for conversion to a preliminary set of V/STOL requirements using a broad background of flight experience and pilots' comments from VTOL and STOL aircraft, BLC (boundary-layer-control) equipped aircraft, variable stability aircraft, flight simulators and landing approach studies. The report contains a discussion of the reasoning behind and the sources of information leading to suggested requirements. The results of the study indicate that the majority of V/STOL requirements can be defined by modifications to the helicopter and/or airplane requirements by appropriate definition of reference speeds. Areas where a requirement is included but where the information is felt to be inadequate to establish a firm quantitative requirement include the following: Control power and damping relationships about all axes for various sizes and types of aircraft; control power, sensitivity, d-amping and response for height control; dynamic longitudinal and dynamic lateral- directional stability in the transition region, including emergency operation; hovering steadiness; acceleration and deceleration in transition; descent rates and flight-path angles in steep approaches, and thrust margin for approach.

Author

Aircraft Control; V/STOL Aircraft; Quality; Aircraft Stability; Control Stability; Controllability; Aerodynamics

19980223625 **NASA Langley Research Center, Hampton, VA USA**

Summary and Analysis of Horizontal-Tail Contribution to Longitudinal Stability of Swept-Wing Airplanes at Low Speeds

Neely, Robert H., NASA Langley Research Center, USA; Griner, Roland F., NASA Langley Research Center, USA; 1959; 96p; In English

Report No.(s): NASA-TR-R-49; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Air-flow characteristics behind wings and wing-body combinations are described and are related to the downwash at specific tail locations for unseparated and separated flow conditions. The effects of various parameters and control devices on the air-flow characteristics and tail contribution are analyzed and demonstrated. An attempt has been made to summarize certain data by empir-

ical correlation or theoretical means in a form useful for design. The experimental data herein were obtained mostly at Reynolds numbers greater than 4 x 10(exp 6) and at Mach numbers less than 0.25.
Author
Body-Wing Configurations; Longitudinal Stability; Swept Wings; Separated Flow; Control Equipment

19980223923 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA
Nonlinear Adaptive Flight Control with a Backstepping Design Approach
Steinberg, Marc L., Naval Air Warfare Center, USA; Page, Anthony B., Naval Air Warfare Center, USA; Jan. 1998; 12p; In English
Report No.(s): AD-A350986; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper examines the use of adaptive backstepping for multi-axis control of a high performance aircraft. The control law is demonstrated on a 6 Degree-of-Freedom simulation with nonlinear aerodynamic and engine models, actuator models with saturation, and turbulence. Simulation results are demonstrated for large pitch-roll maneuvers, and for maneuvers with failure of the right stabilator. There are substantial differences between the control law design and simulation models, which are used to demonstrate some robustness aspects of this control law. Actuator saturation is shown to be a considerable problem for this type of controller. However, the flexibility of the backstepping design provides opportunities for improvement. In particular, the Lyapunov function is modified so that the growth of integrated error and the rate of change of parameter growth are both reduced when the surface commands are growing at a rate that will likely saturate the actuators. In addition, the deadzone technique from robust linear adaptive control is applied to improve robustness to turbulence.
DTIC
Control Theory; Controllers; Supersonic Aircraft; Aircraft Models; Flight Control; Adaptive Control

19980223967 NASA Langley Research Center, Hampton, VA USA
Performance, Stability, and Control Investigation at Mach Numbers from 0.60 to 1.05 of a Model of the "Swallow" with Outer Wing Panels Swept 75 degree with and without Power Simulations
Schmeer, James W., NASA Langley Research Center, USA; Cassetti, Marlowe D., NASA Langley Research Center, USA; Jun. 23, 1960; 66p; In English
Report No.(s): NASA-TM-SX-306; L-1014; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation of the performance, stability, and control characteristics of a variable-sweep arrow-wing model with the outer wing panels swept 75 deg. has been conducted in the Langley 16-foot transonic tunnel. Four outboard engines located above and below the wing provided propulsive thrust, and, by deflecting in the pitch direction and rotating in the lateral plane, also produced control forces. The engine nacelles incorporated swept lateral and vertical fins for aerodynamic stability and control. Jet-off data were obtained with flow-through nacelles, simulating inlet flow; jet thrust and hot-jet interference effects were obtained with faired-nose nacelles housing hydrogen peroxide gas generators. Six-component force and moment data were obtained at Mach numbers from 0.60 to 1.05 through a range of angles of attack and angles of side-slip. Control characteristics were obtained by deflecting the nacelle-fin combinations as elevators, rudders, and ailerons at several fixed angles for each control. The results indicate that the basic wing-body configuration becomes neutrally stable or unstable at a lift coefficient of 0.15; addition of nacelles with fins delayed instability to a lift coefficient of 0.30. Addition of nacelles to the wing-body configuration increased minimum drag from 0.0058 to 0.0100 at a Mach number of 0.60 and from 0.0080 to 0.0190 at a Mach number of 1.05 with corresponding reductions in maximum lift-drag ratio of 12 percent and 33 percent, respectively. The nacelle-fin combinations were ineffective as longitudinal controls but were adequate as directional and lateral controls. The model with nacelles and fins was directionally and laterally stable; the stability generally increased with increasing lift. Jet interference effects on stability and control characteristics were small but the adverse effects on drag were greater than would be expected for isolated nacelles.
Author
Aerodynamic Stability; Body-Wing Configurations; Aerodynamic Coefficients; Nacelles; Arrow Wings; Control Surfaces; Directional Control; Transonic Speed

19980223987 Naval Air Warfare Center, Aircraft Div., Patuxent River, MD USA
Robust Command Augmentation System Design Using Genetic Methods
Sweriduk, G. D., Optimal Synthesis, USA; Menon, P. K., Optimal Synthesis, USA; Stienberg, M. L., Naval Air Warfare Center, USA; Jan. 1998; 9p; In English
Report No.(s): AD-A350849; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

This paper describes the use of a genetic search method in the design of a command augmentation system for a high-performance aircraft. A genetic algorithm is used in the design of H(infinity) controllers for the longitudinal and lateral-directional channels by selecting the weighting functions. The integral of absolute value of error between the actual response and that of an ideal model is used as the fitness criterion, along with additional terms to penalize for cross-coupling between Ps and ny; non-minimum phase behavior, and the closed-loop infinity-norm bound, gamma. Starting from an initial population of weighting functions, the algorithm generates new functions with the goal of improving the fitness. These controllers are then evaluated in a 6 degree-of-freedom nonlinear model of the aircraft.

DTIC

Genetic Algorithms; Feedback Control; Control Systems Design; Computer Aided Design; Control Theory; Flight Control

19980227076 **NASA Lewis Research Center, Cleveland, OH USA**

Static Stability and Control of Canard Configurations at Mach Numbers from 0.70 to 2.22 - Triangular Wing and Canard with Twin Vertical Tails

Peterson, Victor L., NASA Lewis Research Center, USA; Jun. 1961; 40p; In English

Report No.(s): NASA-TN-D-1033; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The static aerodynamic characteristics of a canard airplane configuration having twin vertical stabilizing surfaces are presented. The model consisted of a wing and canard both of triangular plan form and aspect ratio 2 mounted on a Sears-Haack body of fineness ratio 12.5 and two swept and tapered wing-mounted vertical tails of aspect ratio 1.35. Data are presented for Mach numbers from 0.70 to 2.22 and for angles of attack from -6 to +18 deg. at 0 and 5 deg. sideslip. Tests were made with the canard off and with the canard on. Nominal canard deflection angles ranged from 0 to 10 deg. The Reynolds number was 3.68 x 10(exp 6) based on the wing mean aerodynamic chord. Selected portions of the data obtained in this investigation are compared with previously published results for the same model having a single vertical tail instead of twin vertical tails. Without the canard, the directional stability at supersonic Mach numbers and high angles of attack was improved slightly by replacing the single tail with twin tails. However, at a Mach number of 0.70, the directional stability of the twin-tail model deteriorated rapidly with increasing angle of attack above 10 deg. and fell considerably below the level for the single-tail model. At subsonic speeds the directional stability of the twin-tail model with the canard was comparable to that for the single-tail model and at supersonic speed it was considerably greater at high angles of attack. Unlike the single-tail model, the twin-tail model at 50 sideslip exhibited an unstable break in the variation of pitching-moment coefficient with lift coefficient near 10 deg. angle of attack for 0.70 Mach number.

Author

Static Aerodynamic Characteristics; Canard Configurations; Directional Stability; Static Stability; Subsonic Speed; Tail Assemblies; Aerodynamic Coefficients; Delta Wings

19980227082 **NASA Ames Research Center, Moffett Field, CA USA**

The Effect of Lateral-Directional Control Coupling on Pilot Control of an Airplane as Determined in Flight and in a Fixed-Base Flight Simulator

Vomaske, Richard F., NASA Ames Research Center, USA; Sadoff, Melvin, NASA Ames Research Center, USA; Drinkwater, Fred J., III, NASA Ames Research Center, USA; Nov. 1961; 46p; In English

Report No.(s): NASA-TN-D-1141; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight and fixed-base simulator study was made of the effects of aileron-induced yaw on pilot opinion of aircraft lateral-directional controllability characteristics. A wide range of adverse and favorable aileron-induced yaw was investigated in flight at several levels of Dutch-roll damping. The flight results indicated that the optimum values of aileron- induced yaw differed only slightly from zero for Dutch-roll damping from satisfactory to marginally controllable levels. It was also shown that each range of values of aileron-induced yawing moment considered satisfactory, acceptable, or controllable increased with an increase in the Dutch- roll damping. The increase was most marked for marginally controllable configurations exhibiting favorable aileron-induced yaw. Comparison of fixed-base flight simulator results with flight results showed agreement, indicating that absence of kinesthetic motion cues did not markedly affect the pilots' evaluation of the type of control problem considered in this study. The results of the flight study were recast in terms of several parameters which were considered to have an important effect on pilot opinion of lateral-directional handling qualities, including the effects of control coupling. Results of brief tests with a three-axis side-arm controller indicated that for control coupling problems associated with highly favorable yaw and cross-control techniques, use of the three-axis controller resulted in a deterioration of control relative to results obtained with the conventional center stick and rudder pedals.

Author

Directional Control; Lateral Control; Flight Simulators; Controllability; Pilot Induced Oscillation; Aircraft Stability; Aircraft Control

19980227088 **NASA Langley Research Center, Hampton, VA USA**

Dynamic Longitudinal and Directional Stability Derivatives for a 45 deg. Sweptback-Wing Airplane Model at Transonic Speeds

Bielat, Ralph P., NASA Langley Research Center, USA; Wiley, Harleth G., NASA Langley Research Center, USA; Aug. 1959; 54p; In English

Report No.(s): NASA-TM-X-39; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation was made at transonic speeds to determine some of the dynamic stability derivatives of a 45 deg. sweptback-wing airplane model. The model was sting mounted and was rigidly forced to perform a single-degree-of-freedom angular oscillation in pitch or yaw of +/- 2 deg. The investigation was made for angles of attack alpha, from -4 deg. to 14 deg. throughout most of the transonic speed range for values of reduced-frequency parameter from 0.015 to 0.040 based on wing mean aerodynamic chord and from 0.04 to 0.14 based on wing span. The results show that reduced frequency had only a small effect on the damping-in-pitch derivative and the oscillatory longitudinal stability derivative for all Mach numbers M and angles of attack with the exception of the values of damping coefficient near M = 1.03 and alpha = 8 deg. to 14 deg. In this region, the damping coefficient changed rapidly with reduced frequency and negative values of damping coefficient were measured at low values of reduced frequency. This abrupt variation of pitch damping with reduced frequency was a characteristic of the complete model or wing-body-vertical-tail combination. The damping-in-pitch derivative varied considerably with alpha and M for the horizontal-tail-on and horizontal-tail-off configurations, and the damping was relatively high at angles of attack corresponding to the onset of pitch-up for both configurations. The damping-in-yaw derivative was generally independent of reduced frequency and M at alpha = -4 deg. to 4 deg. At alpha = 8 deg. to 14 deg., the damping derivative increased with an increase in reduced frequency and alpha for the configurations having the wing, whereas the damping derivative was either independent of or decreased with increase in reduced frequency for the configuration without the wing. The oscillatory directional stability derivative for all configurations generally decreased with an increase in the reduced-frequency parameter, and, in some instances, unstable values were measured for the model configuration with the horizontal tail removed.

Author

Aircraft Models; Directional Stability; Longitudinal Stability; Transonic Speed; Wind Tunnel Tests; Sweptback Wings; Aerodynamic Configurations; Aircraft Control

19980227090 **NASA Ames Research Center, Moffett Field, CA USA**

A Study of Longitudinal Control Problems at Low and Negative Damping and Stability with Emphasis on Effects of Motion Cues

Sadoff, Melvin, NASA Ames Research Center, USA; McFadden, Norman M., NASA Ames Research Center, USA; Heinle, Donovan R., NASA Ames Research Center, USA; Jan. 1961; 54p; In English; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

As part of a general investigation to determine the effects of simulator motions on pilot opinion and task performance over a wide range of vehicle longitudinal dynamics, a cooperative NASA-AMAL program was conducted on the centrifuge at Johnsville, Pennsylvania. The test parameters and measurements for this program duplicated those of earlier studies made at Ames Research Center with a variable-stability airplane and with a pitch-roll chair flight simulator. Particular emphasis was placed on the minimum basic damping and stability the pilots would accept and on the minimum dynamics they considered controllable in the event of stability-augmentation system failure. Results of the centrifuge-simulator program indicated that small positive damping was required by the pilots over most of the frequency range covered for configurations rated acceptable for emergency conditions only (e.g., failure of a pitch damper). It was shown that the pilot's tolerance for unstable dynamics was dependent primarily on the value of damping. For configurations rated acceptable for emergency operation only, the allowable instability and damping corresponded to a divergence time to double amplitude of about 1 second. Comparisons were made of centrifuge, pitch-chair and fixed-cockpit simulator tests with flight tests. Pilot ratings indicated that the effects of incomplete or spurious motion cues provided by these three modes of simulation were important only for high-frequency, lightly damped dynamics or unstable, moderately damped dynamics. The pitch- chair simulation, which provided accurate angular-acceleration cues to the pilot, compared most favorably with flight. For the centrifuge simulation, which furnished accurate normal accelerations but spurious pitching and longitudinal accelerations, there was a deterioration of pilots' opinion relative to flight results. Results of simulator studies with an analog pilot replacing the human pilot illustrated the adaptive capability of human pilots in coping with the wide range of vehicle dynamics and the control problems covered in this study. It was shown that pilot-response characteristics, deduced by the analog-pilot method, could be related to pilot opinion. Possible application of these results for predicting flight-control problems was illustrated by means of an example control-problem analysis. The results of a brief evaluation of a pencil-type side-arm controller in the centrifuge showed a considerable improvement in the pilots' ability to cope with high-frequency, low-damping dynamics, compared to results obtained with the center stick. This improvement with the pencil controller was attributed

primarily to a marked reduction in the adverse effects of large and exaggerated pitching and longitudinal accelerations on pilot control precision.
Author
Longitudinal Control; Cockpit Simulators; Flight Simulators; Pilot Performance; Flight Control; Dynamic Control; Cues; Flight Tests; Centrifuges

19980227095 NASA Langley Research Center, Hampton, VA USA
Flight Investigation of an Automatic Pitchup Control
Hurt, George J., Jr., NASA Langley Research Center, USA; Whitten, James B., NASA Langley Research Center, USA; Aug. 1960; 30p; In English
Report No.(s): NASA-TN-D-114; L-679; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation of an automatic pitchup control has been conducted by the National Aeronautics and Space Administration at the Langley Research Center. The pitching-moment characteristics of a transonic fighter airplane which was subject to pitchup were altered by driving the stabilizer in accordance with a signal that was a function of a combination of the measured angle of attack and the pitching velocity. An angle-of-attack threshold control was used to preset the angle of attack at which the automatic pitchup-control system would begin to drive the stabilizer. No threshold control as such existed for the pitching-velocity signal. A summing linkage in series with the pilot's longitudinal control allowed the automatic pitchup-control system to drive the stabilizer 13.5 percent of the total stabilizer travel independently of the pilot's control. Tests were made at an altitude of 35,000 feet over a Mach number range of 0.80 to 0.90. Various gearings between the control and the sensing devices were investigated. The automatic system was capable of extending the region of positive stability for the test airplane to angles of attack above the basic-airplane pitchup threshold angle of attack. In most cases a limit-cycle oscillation about the airplane pitch axis occurred.
Author
Pitching Moments; Fighter Aircraft; Automatic Control; Stability Tests; Longitudinal Control; Aircraft Design; Transonic Flight

19980227100 NASA Langley Research Center, Hampton, VA USA
An Experimental Investigation of the Effects of Mach Number, Stabilizer Dihedral, and Fin Torsional Stiffness on the Transonic Flutter Characteristics of a Tee-Tail
Land, Norman S., NASA Langley Research Center, USA; Fox, Annie G., NASA Langley Research Center, USA; Oct. 1961; 26p; In English
Report No.(s): NASA-TN-D-924; L-1611; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A transonic flutter investigation was made of elastically and dynamically scaled models of the tee-tail of a patrol bomber. It was found that removal of the 15 deg. dihedral of the stabilizer used on the airplane raised the flutter boundary to higher dynamic pressures. The effect of Mach number on the flutter boundary was different for dihedral angles of 0 and 15 deg. The dynamic pressure at the flutter boundary increased approximately linearly with the torsional stiffness of the fin. High-speed motion pictures indicated that the flutter mode consisted primarily of fin bending and fin torsion.
Author
Transonic Flutter; Flutter Analysis; Tail Assemblies; Bomber Aircraft; Mach Number; T Tail Surfaces; Horizontal Tail Surfaces; Stabilizers (Fluid Dynamics); Fins

19980227183 NASA Langley Research Center, Hampton, VA USA
Transonic and Supersonic Flutter Investigation of 1/2-Size Models of All-Movable Canard Surface of an Expendable Powered Target
Ruhlin, Charles L., NASA Langley Research Center, USA; Tuovila, W. J., NASA Langley Research Center, USA; 1961; 40p; In English
Report No.(s): NASA-TM-SX-616; L-1303; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A transonic and a supersonic flutter investigation of 1/2-size models of the all-movable canard surface of an expendable powered target has been conducted in the Langley transonic blowdown tunnel and in the Langley 9- by 18-inch supersonic aeroelasticity tunnel, respectively. The transonic investigation covered a Mach number range from 0.7 to i.3, and the supersonic investigation was made at Mach numbers 1.3, 2.O, and 2.55. The effects on the flutter characteristics of the models of different levels of stiffness and of free play in the pitch control linkage were examined. The semispan models, which were tested at an angle of attack of 0 deg, had pitch springs with the scaled design and 1/2 the scaled design pitch stiffness and total free play in pitch ranging from 0 to 1 deg. An additional model configuration which had a pitch spring 1/4 the scaled design pitch stiffness and no free play in pitch was included in the supersonic tests. All model configurations investigated were flutter free up to dynamic pressures 32 percent

greater than those required for flight throughout the Mach number range. Several model configurations were tested to considerably higher dynamic pressures without obtaining flutter at both transonic and supersonic speeds.
Author
Flutter Analysis; Wind Tunnel Tests; Semispan Models; Wind Tunnel Models; Canard Configurations; Transonic Speed; Supersonic Speed; Aeroelasticity; Aircraft Structures

19980227306 **NASA Ames Research Center, Moffett Field, CA USA**
A Self-Adaptive Missile Guidance System for Statistical Inputs
Peery, H. Rodney, NASA Ames Research Center, USA; Nov. 1960; 34p; In English
Report No.(s): NASA-TN-D-343; A-400; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method of designing a self-adaptive missile guidance system is presented. The system inputs are assumed to be known in a statistical sense only. Newton's modified Wiener theory is utilized in the design of the system and to establish the performance criterion. The missile is assumed to be a beam rider, to have a g limiter, and to operate over a flight envelope where the open-loop gain varies by a factor of 20. It is shown that the percent of time that missile acceleration limiting occurs can be used effectively to adjust the coefficients of the Wiener filter. The result is a guidance system which adapts itself to a changing environment and gives essentially optimum filtering and minimum miss distance.
Author
Missiles; Guidance (Motion); Missile Control; Design Analysis; Trajectory Control

19980227331 **West Virginia Univ., Morgantown, WV USA**
Neural Network Autopilot System for a Mathematical Model of the Boeing 747
Cottrill, Gerald C.; Aug. 04, 1998; 158p; In English
Report No.(s): AD-A350857; Rept-98-042; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

Artificial neural networks can be defined as approximate mathematical models of the human brain's learning activities. In recent years neural networks have demonstrated abilities to perform autopilot and fault tolerant control tasks when applied to non-linear numerical aircraft simulations. Five on-line learning neural network autopilot systems, trained with the Standard and Extended Back-Propagation algorithms, were applied to a six degree-of-freedom non-linear simulation of a Boeing 747-200. The performance of the autopilots was compared based on their abilities to perform maneuvers at linear conditions and to adapt at non-linear conditions to restore steady state conditions. Linear maneuvers were performed by introducing reference values of altitude and speed, pitch angle, roll angle, or heading angle. The performance using the SBPA was satisfactory, but the EBPA performance was clearly superior throughout the entire range maneuvers while compensating for lightly damped phugoid and Dutch roll modes. Non-linear adaptation investigations were performed by exciting the non-linear terms in the equations of motion. The non-linear conditions were achieved in two ways: by simultaneously exciting pitch and roll rates with maximum elevator and aileron inputs, and the other by simultaneously exciting roll, pitch, and yaw rates with maximum elevator, aileron, and rudder inputs. The EBPA based controllers were able to regain steady state conditions for both non-linear tests with better transient performance than their SBPA counterparts. The SBPA showed only limited ability to adapt in cases where all three angular rates were excited. Artificial neural networks trained on-line using the Extended Back-Propagation algorithm are concluded to be better suited for autopilot systems for the 1/25 scale Boeing 747 based on their superior abilities to perform linear maneuvers and regain steady state conditions when at non-linear conditions.
DTIC
Automatic Pilots; Boeing 747 Aircraft; Mathematical Models; Neural Nets; On-Line Systems; Nonlinear Systems; Artificial Intelligence; Maneuvers

19980227333 **NASA Langley Research Center, Hampton, VA USA**
Theoretical Analysis of the Longitudinal Behavior of an Automatically Controlled Supersonic Interceptor During the Attack Phase
Gates, Ordway B., Jr., NASA Langley Research Center, USA; Woodling, C. H., NASA Langley Research Center, USA; 1959; 26p; In English
Report No.(s): NASA-TR-R-19; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Theoretical analysis of the longitudinal behavior of an automatically controlled supersonic interceptor during the attack phase against a nonmaneuvering target is presented. Control of the interceptor's flight path is obtained by use of a pitch rate command

system. Topics lift, and pitching moment, effects of initial tracking errors, discussion of normal acceleration limited, limitations of control surface rate and deflection, and effects of neglecting forward velocity changes of interceptor during attack phase.
Author
Flight Paths; Deflection; Pitching Moments; Control Surfaces

19980227351 NASA Langley Research Center, Hampton, VA USA

Effects of Control-Response Characteristics on the Capability of Helicopter for Use as a Gun Platform

Pegg, Robert J., NASA Langley Research Center, USA; Connor, Andrew B., NASA Langley Research Center, USA; Sep. 1960; 20p; In English
Report No.(s): NASA-TN-D-464; L-796; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation with a variable-stability helicopter was undertaken to ascertain the steadiness and ability to "hold on" to the target of a helicopter employed as a gun platform. Simulated tasks were per formed under differing flight conditions with the control-response characteristics of the helicopter varied for each task. The simulated gun-platform mission included: Variations of headings with respect to wind, constant altitude and "swing around" to a wind heading of 0 deg, and increases in altitude while performing a swing around to a wind heading of 0 deg. The results showed that increases in control power and damping increased pilot ability to hold on to the target with fewer yawing oscillations and in a shorter time. The results also indicated that wind direction must be considered in accuracy assessment. Greatest accuracy throughout these tests was achieved by aiming upwind.
Author
Helicopters; Wind Direction; Damping; Flight Conditions; Targets

19980227363 NASA Langley Research Center, Hampton, VA USA

Transonic Flutter Characteristics of a 45 deg Sweptback Wing with Various Distributions of Ballast Along the Leading Edge

Unangst, John R., NASA Langley Research Center, USA; Dec. 1959; 32p; In English
Report No.(s): NASA-TM-X-135; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the use of ballast at the leading edge of a sweptback wing as a flutter fix has been made. The investigation was conducted in the Langley transonic blowdown tunnel with wing models which had an aspect ratio of 4, sweepback of the quarter-chord line of 450, and a taper ratio of 0.2. Four ballast configurations, which included different amounts of ballast distributed at two different span-wise locations, were investigated. Full-span sting-mounted models were employed. Data were obtained over a Mach number range from 0.65 to 1.32. Comparison of the data for the ballasted wings with data for a similar wing without ballast shows that in the often critical Mach number range between 0.85 and 1.05, the dynamic pressure required for flutter is increased by as much as 100 percent due to the addition of about 6 percent of the wing mass as ballast at the leading edge of the outboard sections. Furthermore, there are indications that similar benefits of leading-edge ballast can be obtained at Mach numbers above M = 1.1. Changing the spanwise location of the ballast and increasing the amount of the ballast by a factor of about 2 had very little additional effect on the dynamic pressure required for flutter. The possibility, therefore, exists that the beneficial effects obtained may be accomplished by using less than the minimum of about 6 percent of the wing mass as ballast as investigated in this paper.
Author
Transonic Flutter; Flutter Analysis; Critical Velocity; Sweptback Wings; Dynamic Pressure; Aspect Ratio

19980227404 NASA Langley Research Center, Hampton, VA USA

Low-Subsonic Measurements of the Static Stability and Control and Oscillatory Stability Derivatives of a Proposed Reentry Vehicle Having an Extensible Heat Shield for High-Drag Reentry

Johnson, Joseph L., Jr., NASA Langley Research Center, USA; Boisseau, Peter C., NASA Langley Research Center, USA; Aug. 1961; 38p; In English
Report No.(s): NASA-TN-D-892; L-1329; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A low-speed investigation has been made to determine the static and oscillatory longitudinal and lateral stability derivatives of a proposed reentry vehicle having an extensible heat shield for reentry at high angles of attack. The heat shield is extended forward to give the desired aerodynamic-center position for high-angle-of-attack reentry and, after completion of the reentry phase, is retracted to give stability and trim for gliding flight at low angles of attack. Near an angle of attack of 900 the reentry configuration was statically stable both longitudinally and directionally, had positive dihedral effect, and had positive damping

in roll but zero damping in yaw. The landing configuration had positive damping in pitch, roll, and yaw over the test angle-of-attack range but was directionally unstable and had negative dihedral effect between an angle of attack of about 10 and 20 deg.
Author
Longitudinal Stability; Reentry Vehicles; Static Stability; Reentry; Aerodynamic Balance; Heat Shielding; Lateral Stability; Subsonic Speed

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19980227742 NASA Langley Research Center, Hampton, VA USA

Investigation of Longitudinal and Lateral Stability Characteristics of a Six-Propeller Deflected-Slipstream VTOL Model with Boundary-Layer Control Including Effects of Ground Proximity

Grunwald, Kalman J., NASA Langley Research Center, USA; Jan. 1961; 94p; In English
Report No.(s): NASA-TN-D-445; L-951; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An investigation of the longitudinal and lateral stability and control and Performance characteristics of a six-propeller deflected- slipstream vertical-take-off-and-landing (VTOL) model in the transition speed range was conducted in the 17-foot test section of the Langley 300-MPH 7- by 10-foot tunnel. A complete analysis of the data was not conducted. A modest amount of blowing boundary-layer control was necessary to achieve transition without wing stall.
Author
Lateral Stability; Longitudinal Stability; Wind Tunnel Tests; Vertical Takeoff Aircraft; Boundary Layer Control; Propeller Slipstreams; Aircraft Configurations

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19980227758 NASA Langley Research Center, Hampton, VA USA

Stability and Control Characteristics of a Model of an Aerial Vehicle Supported by Four Ducted Fans

Parlett, Lysle P., NASA Langley Research Center, USA; Aug. 1961; 20p; In English
Report No.(s): NASA-TN-D-937; L-1482; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The stability and control characteristics of a simple, lightly loaded model approximately one-third the size of a full-scale vehicle have been investigated by a series of free-flight tests. The model is representative of a type of vertically rising aircraft which would utilize four ducted fans as its sole source of lift and propulsion. The ducts were arranged in a rectangular pattern and were fixed to the airframe so that their axes of revolution were vertical for hovering flight. Control moments were provided by remotely controlled compressed-air jets at the sides and ends of the model. In hovering, the model in its original configuration exhibited divergent oscillations about both the roll and pitch axes. Because these oscillations were of a rather short period., the model was very difficult to control by the use of remote controls only. The model could be completely stabilized by the addition of a sufficient amount of artificial damping. The pitching oscillation was made easier to control by increasing the distance between the forward and rearward pairs of ducts. In forward flight, with the model in its original configuration, the top speed was limited by the development of an uncontrollable pitch-up. Large forward tilt angles were required for trim at the highest speeds attained. With the model rotated so that the shorter axis became the longitudinal axis, the pitch trim problem was found to be less than with the longer axis as the longitudinal axis. The installation of a system of vanes in the slipstream of the forward ducts reduced the tilt angle but increased the power required.
Author
Wind Tunnel Tests; Ducted Fans; Flight Characteristics; Aerodynamic Configurations; Wind Tunnel Models; Aerodynamic Stability; Aircraft Control; Lift Fans; Vertical Takeoff

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19980227762 NASA Langley Research Center, Hampton, VA USA

Dynamic Stability and Control Problems of Piloted Reentry from Lunar Missions

Moul, Martin T., NASA Langley Research Center, USA; Schy, Albert A., NASA Langley Research Center, USA; Williams, James L., NASA Langley Research Center, USA; Nov. 1961; 22p; In English
Report No.(s): NASA-TN-D-986; L-1764; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A fixed-base simulator investigation has been made of stability and control problems during piloted reentry from lunar missions. Reentries were made within constraints of acceleration and skipping, in which the pilot was given simulated navigation tasks of altitude and heading angle commands. Vehicles considered included a blunt-face, high-drag capsule, and a low-drag lifting cone, each of which had a trim lift-drag ratio of 0.5. With the provision of three-axis automatic damping, both vehicles were easily controlled through reentry after a brief pilot-training period. With all dampers out, safe reentries could be made and both

vehicles were rated satisfactory for emergency operation. In damper-failure conditions resulting in inadequate Dutch roll damping, the lifting-cone vehicle exhibited control problems due to excessive dihedral effect and oscillatory acceleration effects.
Author

Manned Reentry; Aerodynamic Stability; Altitude Simulation; Lateral Stability; Dynamic Stability; Lifting Reentry Vehicles; Lunar Exploration

19980227771 NASA Langley Research Center, Hampton, VA USA

Lateral Stability and Control Characteristics of a Four-Propeller Deflected-Slipstream VTOL Model Including the Effects of Ground Proximity

Kuhn, Richard E., NASA Langley Research Center, USA; Grunwald, Kalman J., NASA Langley Research Center, USA; Jan. 1961; 72p; In English

Report No.(s): NASA-TN-D-444; L-895; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The investigation of the lateral-directional stability and control characteristics of a four-propeller deflected-slipstream VTOL model in the transition speed range was conducted in the 17-foot test section of the Langley 300-MPH 7- by 10-foot tunnel. A large fairing on top of the rear fuselage was needed to eliminate directional instability in the power-off flaps-retracted condition. Even with this fairing some instability at small sideslip angles remained for power-on conditions with low flap deflections. The configuration exhibited a high level of dihedral effect which, coupled with the directional instability, will probably produce an undesirable Dutch roll oscillation.
Author

Vertical Takeoff Aircraft; Lateral Stability; Lateral Control; Wind Tunnel Tests; Wind Tunnel Models; Propeller Slipstreams; Aircraft Configurations

19980227774 NASA Langley Research Center, Hampton, VA USA

Effect at High Subsonic Speeds of Fuselage Forebody Strakes on the Static Stability and Vertical-Tail-Load Characteristics of a Complete Model Having a Delta Wing

Polamus, Edward C., NASA Langley Research Center, USA; Spreemann, Kenneth P., NASA Langley Research Center, USA; May 1961; 32p; In English

Report No.(s): NASA-TN-D-903; L-1531; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation at high subsonic speeds has been conducted to determine the effect of fuselage forebody strakes on the static stability and the vertical-tail-load characteristics of an airplane-type configuration having a delta wing. The tests were made at Mach numbers from 0.60 to 0.92 corresponding to Reynolds numbers from 3.0 x 10(exp 6) to 4.2 x 10(exp 6), based on the wing mean aerodynamic chord, and at angles of attack from approximately -2 to 24 deg. The strakes provided improvements in the directional stability characteristics of the wing-fuselage configuration which were reflected in the characteristics of the complete configuration in the angle-of-attack range where extreme losses in directional stability quite often occur. It was also found that the strakes, through their beneficial effect on the wing-fuselage directional stability, reduced the vertical-tail load per unit restoring moment at high angles of attack. The results also indicated that, despite the inherent tendency for strakes to produce a pitch-up, acceptable pitching-moment characteristics can be obtained provided the strakes are properly chosen and used in conjunction with a wing-body-tail configuration characterized by increasing stability with increasing lift.
Author

Body-Wing and Tail Configurations; Subsonic Speed; Strakes; Wind Tunnel Tests; Wind Tunnel Models; Forebodies; Fuselages; Directional Stability; Aircraft Stability; Delta Wings; Static Stability

19980227775 NASA Langley Research Center, Hampton, VA USA

Study of an Active Control System for a Spinning Body

Adams, J. J., NASA Langley Research Center, USA; Jun. 1961; 28p; In English

Report No.(s): NASA-TN-D-905; L-1519; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The mission requirements for some satellites require that they spin continuously and at the same time maintain a precise direction of the spin axis. An analog-computer study has been made of an attitude control system which is suitable for such a satellite. The control system provides the necessary attitude control through the use of a spinning wheel, which will provide precession torques, commanded by an automatic closed-loop servomechanism system. The sensors used in the control loop are rate gyroscopes for damping of any wobble motion and a sun seeker for attitude control. The results of the study show that the controller can eliminate the wobble motion of the satellite resulting from a rectangular pulse moment disturbance and then return the spin axis to the reference space axis. The motion is damped to half amplitude in less than one cycle of the wobble motion. The controller can also reduce the motion resulting from a step change in product of inertia both by causing the new principal axis to be steadily

alined with the spin vector and by reducing the cone angle generated by the reference body axis. These methods will reduce the motion whether the satellite is a disk, sphere, or rod configuration.
Author
Active Control; Spin Reduction; Spin Dynamics; Satellite Attitude Control; Controllers; Satellite Rotation

19980227800 NASA Langley Research Center, Hampton, VA USA
Low-Speed Investigation of the Effects of Frequency and Amplitude of Oscillation in Sideslip on the Lateral Stability Derivatives of a 60 deg Delta Wing, a 45 deg Sweptback Wing and an Unswept Wing
Lichtenstein, Jacob H., NASA Langley Research Center, USA; Williams, James L., NASA Langley Research Center, USA; May 1961; 20p; In English
Report No.(s): NASA-TN-D-896; L-1608; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A low-speed investigation has been conducted in the Langley stability tunnel to study the effects of frequency and amplitude of sideslipping motion on the lateral stability derivatives of a 60 deg. delta wing, a 45 deg. sweptback wing, and an unswept wing. The investigation was made for values of the reduced-frequency parameter of 0.066 and 0.218 and for a range of amplitudes from +/- 2 to +/- 6 deg. The results of the investigation indicated that increasing the frequency of the oscillation generally produced an appreciable change in magnitude of the lateral oscillatory stability derivatives in the higher angle-of-attack range. This effect was greatest for the 60 deg. delta wing and smallest for the unswept wing and generally resulted in a more linear variation of these derivatives with angle of attack. For the relatively high frequency at which the amplitude was varied, there appeared to be little effect on the measured derivatives as a result of the change in amplitude of the oscillation.
Author
Wind Tunnel Tests; Delta Wings; Sweptback Wings; Unswept Wings; Stability Derivatives; Lateral Stability; Sideslip; Wing Oscillations

09
RESEARCH AND SUPPORT FACILITIES (AIR)

Includes airports, hangars and runways; aircraft repair and overhaul facilities; wind tunnels; shock tubes; and aircraft engine test stands.

19980223937 Federal Aviation Administration, Technical Center, Atlantic City, NJ USA
Automated Surface Observing System (ASOS) Controller Equipment (ACE) Operational Test and Evaluation *Final Report*
Horan, Colleen, Federal Aviation Administration, USA; Melillo, Michael R., Fed[illegible] Aviation Administration, USA; Peio, Karen J., Federal Aviation Administration, USA; Nuzman, Edward F., Federal Aviation Administration, USA; Vicente, James P., Federal Aviation Administration, USA; May 1998; 69p; In English
Report No.(s): AD-A350596; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The Automated Surface Observing System (ASOS) Controller Equipment (ACE) system is a display system that provides weather products from the ASOS and other weather product systems to the Federal Aviation Administration (FAA) Air Traffic Control Towers (ATCTs), Terminal Radar Approach Control (TRACON), and other selected locations. Operational Test and Evaluation (OT&E) of the ACE was conducted in four phases, commencing at the FAA William J. Hughes Technical Center in July 1995 and concluding at the Will Rogers World Airport, Oklahoma City Oklahoma (OKC) and Dallas/Ft. Worth International Airport, Irving, Texas (DFW), in April 1997. The purpose of the OT&E was to evaluate the performance of the ACE display system. This final report describes the results of OT&E testing conducted on the ACE.
DTIC
Controllers; Evaluation; Display Devices; Information Transfer

19980223963 Veridian, Moffett Field, CA USA
The Real-Time Wall Interference Correction System of the NASA Ames 12-Foot Pressure Wind Tunnel
Ulbrich, Norbert, Veridian, USA; Jul. 1998; 206p; In English
Contract(s)/Grant(s): NAS2-13605
Report No.(s): NASA/CR-1998-208537; A-98-11989; NAS 1.26:208537; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

An improved version of the Wall Signature Method was developed to compute wall interference effects in three-dimensional subsonic wind tunnel testing of aircraft models in real-time. The method may be applied to a full-span or a semispan model. A

simplified singularity representation of the aircraft model is used. Fuselage, support system, propulsion simulator, and separation wake volume blockage effects are represented by point sources and sinks. Lifting effects are represented by semi-infinite line doublets. The singularity representation of the test article is combined with the measurement of wind tunnel test reference conditions, wall pressure, lift force, thrust force, pitching moment, rolling moment, and pre-computed solutions of the subsonic potential equation to determine first order wall interference corrections. Second order wall interference corrections for pitching and rolling moment coefficient are also determined. A new procedure is presented that estimates a rolling moment coefficient correction for wings with non-symmetric lift distribution. Experimental data obtained during the calibration of the Ames Bipod model support system and during tests of two semispan models mounted on an image plane in the NASA Ames 12 ft. Pressure Wind Tunnel are used to demonstrate the application of the wall interference correction method.
Author
Subsonic Flow; Support Systems; Wall Flow; Wall Pressure; Subsonic Wind Tunnels; Force Distribution

19980227544 Pittsburgh State Univ., KS USA
Analysis, Specification and Implementation of an Automated Programmable Control System and Virtual Instrumentation to Improve and Advance the Operation of the Slack Thermal High Vacuum Chamber
Buchanan, Randy K., Pittsburgh State Univ., USA; 1997 Research Reports: NASA/ASEE Summer Faculty Fellowship Program; Dec. 1997, pp. 11-20; In English; Also announced as 19980227542; No Copyright; Avail: CASI; A02, Hardcopy; A03, Microfiche

The Slack Thermal Vacuum Chamber was designed to process spacecraft and ground support equipment for the Materials Science Laboratory at Kennedy Space Center (KSC). The chamber recently became inoperative and was thus identified to be equipped with a modern control system to enable support of the launch of the Space Shuttle, expendable rockets, and their respective payloads. Installation of a modern computerized programmable control system was performed, which included connection of new control hardware and complex programming of the controller. Furthermore, a virtual instrumentation system was created with the use of an additional computer and the incorporation of virtual instrumentation software. This report characterizes the evolution and successful completion of this modernization process.
Author
Control Systems Design; Numerical Control; Vacuum Chambers; Programmable Logic Devices

19980227789 NASA Langley Research Center, Hampton, VA USA
Description of a 2-Foot Hypersonic Facility at the Langley Research Center
Stokes, George M., NASA Langley Research Center, USA; Sep. 1961; 26p; In English
Report No.(s): NASA-TN-D-939; L-1390; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report describes the mechanical and aerodynamic features of a two-foot hypersonic facility at the Langley Research Center. The facility provides for the testing of aerodynamic models in the Mach number range between 3 and 7 at approximate Reynolds numbers between 0.5 x 10(exp 6) and 1.0 x 10(exp 6). The facility was designed to obtain the needed pressure ratio through the use of ejector nozzles. Compressors driving the ejectors operate continuously at a pressure ratio of 4 and thus give the facility a continuous running capability. Curves are presented to show the ranges of total temperature, total pressure, Reynolds number dynamic pressure, and static pressure available in the tunnel. The flow in the test section is suitable for model tests at all Mach numbers between 3 and 7, although the nozzle blocks were contoured for a Mach number of 6.
Author
Supersonic Wind Tunnels; Aerodynamic Characteristics; Supersonic Speed; Hypersonics; Compressors

19980227798 NASA Langley Research Center, Hampton, VA USA
The Development of an 8-inch by 8-inch Slotted Tunnel for Mach Numbers up to 1.28
Little, B. H., Jr., NASA Langley Research Center, USA; Cubbage, James J., Jr., NASA Langley Research Center, USA; Aug. 1961; 14p; In English
Report No.(s): NASA-TN-D-908; L-1005; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An 8-inch by 8-inch transonic tunnel model with test section slotted on two opposite walls was constructed in which particular emphasis -was given to the development of slot geometry, slot-flow reentry section, and short-diffuser configurations for good test-region flow and minimum total-pressure losses. Center-line static pressures through the test section, wall static pressures through the other parts of the tunnel, and total-pressure distributions at the inlet and exit stations of the diffuser were measured- With a slot length equal to two tunnel heights and 1/14 open-area-ratio slotted walls) a test region one tunnel height in length was obtained in which the deviation from the mean Mach number was less than +/- 0.01 up to Mach number 1.15. With 1/7 open-area-ratio slotted walls, a test region 0.84 tunnel heights in length with deviation less than +/- O.01 was obtained up to Mach number 1.26. Increasing the tunnel diffuser angle from 6.4 to 10 deg. increased pressure loss through the tunnel at Mach number 1.20 from

15 percent to 20 percent of the total pressure. The use of other diffusers with equivalent angles of 10 deg. but contoured so that the initial diffusion angle was less than 10 deg. and the final angle was 200 reduced the losses to as low as 16 percent. A method for changing the test-section Mach number rapidly by controlling the flow through a bypass line from the tunnel settling chamber to the slot-flow plenum chamber of the test section was very effective. The test-section Mach number was reduced approximately 5 percent in 1/8 second by bleeding into the test section a flow of air equal to 2 percent of the mainstream flow and 30 percent in 1/4 second with bleed flow equal to 10 percent of the mainstream flow. The rate of reduction was largely determined by the opening rate of the bleed-flow-control valve.
Author
Transonic Wind Tunnels; Slotted Wind Tunnels; Wind Tunnel Apparatus; Pressure Distribution

10
ASTRONAUTICS

Includes astronautics (general); astrodynamics; ground support systems and facilities (space); launch vehicles and space vehicles; space transportation; space communications, spacecraft communications, command and tracking; spacecraft design, testing and performance; spacecraft instrumentation; and spacecraft propulsion and power.

19980221813 **Smithsonian Astrophysical Observatory, Cambridge, MA USA**
In-Space Transportation with Tethers *Annual Report No. 2, 1 Sep. 1997 - 31 Aug. 1998*
Lorenzini, Enrico, Smithsonian Astrophysical Observatory, USA; Estes, Robert D., Smithsonian Astrophysical Observatory, USA; Cosmo, Mario L., Smithsonian Astrophysical Observatory, USA; Aug. 1998; 120p; In English
Contract(s)/Grant(s): NAG8-1303; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

The annual report covers the research conducted on the following topics related to the use of spaceborne tethers for in-space transportation: ProSEDS tether modeling (current collection analyses, influence of a varying tether temperature); proSEDS mission analysis and system dynamics (tether thermal model, thermo-electro-dynamics integrated simulations); proSEDS-tether development and testing (tether requirements, deployment test plan, tether properties testing, deployment tests); and tethers for reboosting the space-based laser (mission analysis, tether system preliminary design, evaluation of attitude constraints).
Author
Tethering; Air Transportation; Models; Dynamic Characteristics; Manufacturing; Design Analysis

19980223079 **NASA Langley Research Center, Hampton, VA USA**
Trajectory Control for Vehicles Entering The Earth's Atmosphere at Small Flight-Path Angles
Eggleston, John M., NASA Langley Research Center, USA; Young, John W., NASA Langley Research Center, USA; 1961; 32p; In English
Report No.(s): NASA-TR-R-89; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Methods of controlling the trajectories of high-drag low-lift vehicles entering the earth's atmosphere at angles of attack near 90 degrees and at initial entry angles up to 3 degrees are studied. The trajectories are calculated for vehicles whose angle of attack can be held constant at some specified value or can be perfectly controlled as a function of some measured quantity along the trajectory. The results might be applied in the design of automatic control systems or in the design of instruments which will give the human pilot sufficient information to control his trajectory properly during an atmospheric entry. Trajectory data are compared on the basis of the deceleration, range, angle of attack, and, in some cases, the rate of descent. The aerodynamic heat-transfer rate and skin temperature of a vehicle with a simple heat-sink type of structure are calculated for trajectories made with several types of control functions.
Author (revised)
Control Systems Design; Trajectory Control; Reentry; Aerodynamic Heat Transfer; Automatic Control; Aerodynamic Heating; Descent Trajectories

19980223473 **NASA Langley Research Center, Hampton, VA USA**
Mars Ascent Vehicle Flight Analysis
Desai, P. N., NASA Langley Research Center, USA; Braun, R. D., NASA Langley Research Center, USA; Engelund, W. C., NASA Langley Research Center, USA; Cheatwood, F. M., NASA Langley Research Center, USA; Kangas, J. A., Jet Propulsion Lab., California Inst. of Tech., USA; 1998; 10p; In English; 7th; Thermophysics and Heat Transfer, 15-18 Jun. 1998, Albuquerque, NM, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA; Original contains color illustrations
Report No.(s): AIAA Paper 98-2850; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The scientific objective of the Mars Surveyor Program 2005 mission is to return Mars rock, soil, and atmospheric samples to Earth for detailed analysis. The present investigation focuses on design of Mars Ascent Vehicle for this mission. Aerodynamic, aerothermodynamic, and trajectory design considerations are addressed to assess the ascent configuration, determine aerodynamic stability, characterize thermal protection system requirements, and ascertain the required system mass. Aerodynamic analysis reveals a subsonic static instability with the baseline configuration; however, stability augmentation options are proposed to mitigate this problem. The ascent aerothermodynamic environment is shown to be benign (on the order of the sea-level boiling point of water on Earth). As a result of these low thermal and pressure loads, a lightweight, low rigidity material can be employed as the aftbody aerodynamic shroud. The required nominal MAV lift-off mass is 426 kg for a December 2006 equatorial launch into a 300-km circular orbit with 30-degree inclination. Off-nominal aerodynamic and atmospheric conditions are shown to increase this liftoff mass by approximately 10%. Through performance of these analyses, the Mars Ascent Vehicle is deemed feasible with respect to the current mission mass and size constraints.
Author
Planetary Geology; Mission Planning; Mars Surface; Meteorology; Design Analysis; Circular Orbits; Afterbodies; Aerothermodynamics; Aerodynamic Stability; Aerodynamic Characteristics

19980223479 **NASA Marshall Space Flight Center, Huntsville, AL USA**
International Space Station Electrodynamic Tether Reboost Study
Johnson, L., NASA Marshall Space Flight Center, USA; Herrmann, M., NASA Marshall Space Flight Center, USA; Jul. 1998; 40p; In English
Report No.(s): NASA/TM-1998-208538; M-886; NAS 1.15:208538; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The International Space Station (ISS) will require periodic reboost due to atmospheric aerodynamic drag. This is nominally achieved through the use of thruster firings by the attached Progress M spacecraft. Many Progress flights to the ISS are required annually. Electrodynamic tethers provide an attractive alternative in that they can provide periodic reboost or continuous drag cancellation using no consumables, propellant, nor conventional propulsion elements. The system could also serve as an emergency backup reboost system used only in the event resupply and reboost are delayed for some reason.
Author
International Space Station; Aerodynamic Drag; Tetherlines; Tethering; Drag Reduction; Orbit Decay

19980223581 **NASA Lewis Research Center, Cleveland, OH USA**
Experimental Performance of Area Ratio 200, 25 and 8 Nozzles on JP-4 Fuel and Liquid Oxygen Rocket Engine
Lovell, J. Calvin, NASA Lewis Research Center, USA; Samanich, Nick E., NASA Lewis Research Center, USA; Barnett, Donald O., NASA Lewis Research Center, USA; Aug. 1960; 18p; In English
Report No.(s): NASA-TM-X-382; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The performance of an area ratio 200 bell-shaped nozzle, an area ratio 25 bell-shaped nozzle, and an area ratio 8 conic nozzle on a JP-4 fuel and liquid-oxygen rocket engine has been determined. Tests were conducted using a nominal 4000-pound-thrust rocket in the Lewis 10- by 10-foot supersonic tunnel, which provided the altitude environment needed for fully expanded nozzle flow. The area ratio 200 nozzle had a vacuum thrust coefficient of 1.96, compared with 1.82 and 1.70 for the area ratio 25 and 8 nozzles, respectively. These values are approximately equal to those for theoretical frozen expansion. The measured value of vacuum specific impulse for the area ratio 200 nozzle was 317 seconds for a combustion-chamber characteristic velocity of 5200 feet per second. The vacuum-specific-impulse increase for the area-ratio increase from 8 to 200 was 46 seconds.
Author
Nozzle Design; Nozzle Flow; Nozzle Geometry; JP-4 Jet Fuel; Conics; Specific Impulse; Thrust

19980223622 **NASA Ames Research Center, Moffett Field, CA USA**
An Approximate Analytical Method for Studying Entry into Planetary Atmospheres
Chapman, Dean R., NASA Ames Research Center, USA; 1959; 48p; In English
Report No.(s): NASA-TR-R-11; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The pair of motion equations for entry into a planetary atmosphere is reduced to a single, ordinary, nonlinear differential equation of second order by disregarding two relatively small terms and by introducing a certain mathematical transformation. The reduced equation includes various terms, certain of which represent the gravity force, the centrifugal acceleration, and the lift force. If these particular terms are disregarded, the differential equation is linear and yields precisely the solution of Allen and Eggers applicable to ballistic entry at relatively steep angles of descent. If all the other terms in the basic equation are disregarded (corresponding to negligible vertical acceleration and negligible vertical component of drag force), the resulting truncated differential equation yields the solution of Sanger for equilibrium flight of glide vehicles with relatively large lift-drag ratios. A number of solutions for lifting and nonlifting vehicles entering at various initial angles also have been obtained from the complete nonlinear equation. These solutions are universal in the sense that a single solution determines the motion and heating of a vehicle of arbitrary weight, dimensions, and shape entering an arbitrary planetary atmosphere. One solution is required for each lift-drag ratio. These solutions are used to study the deceleration, heating rate, and total heat absorbed for entry into Venus, Earth, Mars, and Jupiter. From the equations developed for heating rates, and from available information on human tolerance limits to acceleration stress, approximate conditions for minimizing the aerodynamic heating of a trimmed vehicle with constant lift-drag ratio are established for several types of manned entry.

Author

Planetary Atmospheres; Atmospheric Entry; Aerodynamic Heating; Equations of Motion; Nonlinear Equations; Design Analysis

19980223952 NASA Ames Research Center, Moffett Field, CA USA

A Flight Study of a Power-Off Landing Technique Applicable to Re-Entry Vehicles

Bray, Richard S., NASA Ames Research Center, USA; Drinkwater, Fred J., NASA Ames Research Center, USA; White, Maurice D., NASA Ames Research Center, USA; Jul. 1960; 30p; In English

Report No.(s): NASA-TN-D-323; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A power-off landing technique, applicable to aircraft of configurations presently being considered for manned re-entry vehicles, has been developed and flight tested at Ames Research Center. The flight tests used two configurations of an airplane for which the values of maximum lift-drag ratios were 4.0 and 2.8. Twenty-four idle-power approaches were made to an 8000-foot runway with touchdown point and airspeed accuracies of +/-600 feet and +/-10 knots, respectively. The landing pattern used was designed to provide an explicitly defined flight path for the pilot and, yet, to require no external guidance other than the pilot's view from the cockpit. The initial phase of the approach pattern is a constant high-speed descent from altitude aimed at a ground reference point short of the runway threshold. At a specified altitude and speed, a constant g pull-out is made to a shallow flight path along which the air-plane decelerates to the touchdown point. Repeatability and safety are inherent because of the reduced number of variables requiring pilot judgment, and because of the fact that a missed approach is evident at speeds and altitudes suitable for safe ejection. The accuracy and repeatability of the pattern are indicated by the measured results. The proposed pattern appears to be particularly suitable for configurations having unusual drag variations with speed in the lower speed regime, since the pilot is not required to control speed in the latter portions of the pattern.

Author

Aircraft Configurations; Reentry Vehicles; Flight Tests; Spacecraft Landing

19980223970 NASA Langley Research Center, Hampton, VA USA

Some Effects of Ablation Surface Roughness on the Aerodynamic Characteristics of a Reentry Vehicle at Mach Numbers from 0.30 to 1.00

Decker, John P., NASA Langley Research Center, USA; Abel, Irving, NASA Langley Research Center, USA; Feb. 1971; 80p; In English

Contract(s)/Grant(s): RTOP 124-07-17-07

Report No.(s): NASA-TM-SX-2050; L-7288; AF-AM-833; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been made to determine some of the effects of ablation surface roughness on the aerodynamic characteristics of a reentry vehicle. The vehicle selected for the investigation was the SV-5D/FV-3 PRIME vehicle which is an approximate 0.28-scale model of the X-24A manned low-speed flight research vehicle. The PRIME vehicle was flown on a suborbital flight and subsequently retrieved by a U.S. Air Force cargo airplane. The PRIME vehicle was restored and modified for wind-tunnel testing and some of the effects of ablation surface roughness were determined by testing the ablated model (PRIME vehicle restored and modified for wind-tunnel tests) and a smooth replica. The tests were conducted at Mach numbers from 0.30 to 1.00.

Author

Reentry Vehicles; Suborbital Flight; Surface Roughness Effects; Ablation; Aerodynamic Characteristics; Scale Models; Wind Tunnel Tests; X-24 Aircraft; Aerodynamic Heating

19980223971 NASA Langley Research Center, Hampton, VA USA

Low Subsonic Aerodynamic Characteristics of a Reentry Spacecraft Shape with a High Hypersonic Lift-Drag Ratio

Martin, James A., NASA Langley Research Center, USA; Decker, John P., NASA Langley Research Center, USA; Apr. 1971; 38p; In English

Contract(s)/Grant(s): RTOP 124-07-17-07

Report No.(s): NASA-TM-SX-2097; L-7334; AF-AM-920; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley low-turbulence pressure tunnel to determine the aerodynamic characteristics of a proposed flight-research vehicle with a delta planform and a high hypersonic lift-drag ratio. The tests were conducted at a Mach number of about 0.25, angles of attack from about -9 deg. to 31 deg., angles of sideslip from about -7 deg. to 12 deg., and Reynolds numbers based on body length from 4.5 x 10(exp 6) to 43.2 X 10(exp 6).
Author
Subsonic Speed; Aerodynamic Characteristics; Reentry Vehicles; Wind Tunnel Tests; Aerodynamic Coefficients

19980223977 **NASA Langley Research Center, Hampton, VA USA**
A Concept of a Manned Satellite Reentry Which is Completed with a Glide Landing
Cheatham, Donald C., Compiler, NASA Langley Research Center, USA; Dec. 1959; 46p; In English
Report No.(s): NASA-TM-X-226; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A concept for a manned satellite reentry from a near space orbit and a glide landing on a normal size airfield is presented. The reentry vehicle configuration suitable for this concept would employ a variable geometry feature in order that the reentry could be made at 90 deg. angle of attack and the landing could be made with a conventional glide approach. Calculated results for reentry at a flight-path angle of -1 deg. show that with an accuracy of 1 percent in the impulse of a retrorocket, the desired flight-path angle at reentry can be controlled within 0.02 deg. and the distance traveled to the reentry point, within 100 miles. The reentry point is arbitrarily defined as the point at which the satellite passes through an altitude of about 70 miles. Misalignment of the retrorocket by 10 deg. increased these errors by as much as 0.02 deg. and 500 miles. Intra-atmospheric trajectory calculations show that pure drag reentries starting with flight-path angles of -1 deg. or less produce a peak deceleration of 8g. Lift created by varying the angle of attack between 90 and 60 deg. is effective in decreasing the maximum deceleration and allows the range to the "recovery" point (where transition is made from reentry to gliding flight) to be increased by as much as 2,300 miles. A sideslip angle of 30 deg. allows lateral displacement of the flight path by as much as 60 deg. miles. Reaction controls would provide control-attitude alignment during the orbit phase. For the reentry phase this configuration should have low static longitudinal and roll stability in the 90 deg. angle-of-attack attitude. Control could be effected by leading-edge and trailing-edge flaps. Transition into the landing phase would be accomplished at an altitude of about 100,000 feet by unfolding the outer wing panels and pitching over to low angles of attack. Calculations indicate that glides can be made from the recovery point to airfields at ranges of from 150 to 200 miles, depending upon the orientation with respect to the original course.
Author
Manned Reentry; Angle of Attack; Reentry Vehicles; Glide Landings; Descent Trajectories; Atmospheric Entry; Spacecraft Landing

19980223992 **NASA Ames Research Center, Moffett Field, CA USA**
An Approximate Analytical Method for Studying Atmosphere Entry of Vehicles with Modulated Aerodynamic Forces
Levy, Lionel L., Jr., NASA Ames Research Center, USA; Oct. 1960; 34p; In English
Report No.(s): NASA-TN-D-319; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The dimensionless, transformed, nonlinear differential equation developed in NASA TR R-11 for describing the approximate motion and heating during entry into planetary atmospheres for constant aerodynamic coefficients and vehicle shape has been modified to include entries during which the aerodynamic coefficients and the vehicle shape are varied. The generality of the application of the original equation to vehicles of arbitrary weight, size, and shape and to arbitrary atmospheres is retained. A closed-form solution for the motion, heating, and the variation of drag loading parameter m/C(D)A has been obtained for the case of constant maximum resultant deceleration during nonlifting entries. This solution requires certain simplifying assumptions which do not compromise the accuracy of the results. The closed-form solution has been used to determine the variation of m/C(D)A required to reduce peak decelerations and to broaden the corridor for nonlifting entry into the earth's atmosphere at escape velocity. The attendant heating penalty is also studied.
Author
Atmospheric Entry; Aerospace Vehicles; Aerodynamic Coefficients; Aerodynamic Heating; Aerodynamic Configurations; Differential Equations; Aerodynamic Forces

19980227176 **NASA Langley Research Center, Hampton, VA USA**
Structures for Reentry Heating
Anderson, Roger A., NASA Langley Research Center, USA; Swann, Robert T., NASA Langley Research Center, USA; Sep. 1960; 22p; In English
Report No.(s): NASA-TM-X-313; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The basic structural approaches for dealing with reentry heating of manned vehicles are summarized. The weight and development status of both radiative and ablative shields are given and the application of these shields to various vehicles is indicated.
Author
Aerodynamic Heating; Reentry Effects; Ablation; Spacecraft Construction Materials; Manned Reentry; Temperature Effects

19980227215 NASA Langley Research Center, Hampton, VA USA
Charts Depicting Kinematic and Heating Parameters for a Ballistic Reentry at Speeds of 26,000 to 45,000 Feet Per Second
Lovelace, Uriel M., NASA Langley Research Center, USA; Oct. 1961; 54p; In English
Report No.(s): NASA-TN-D-968; L-1750; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Reentry trajectories, including computations of convective and radiative stagnation-point heat transfer, have been calculated by using equations for a point-mass reentry vehicle entering the atmosphere of a rotating, oblate earth. Velocity was varied from 26,000 to 45,000 feet per second; reentry angle, from the skip limit to -20 deg; ballistic drag parameter, from 50 to 200. Initial altitude was 400,000 feet. Explicit results are presented in charts which were computed for an initial latitude of 38 deg N and an azimuth of 90 deg from north. A method is presented whereby these results may be made valid for a range of initial latitude and azimuth angles.
Author
Radiative Heat Transfer; Aerodynamic Heat Transfer; Reentry Trajectories; Stagnation Point; Drag

19980227432 NASA Langley Research Center, Hampton, VA USA
Preliminary Results on Heat Transfer to the Afterbody of the Apollo Reentry Configuration at a Mach Number of 8
Jones, Robert A., NASA Langley Research Center, USA; Sep. 1962; 14p; In English
Report No.(s): NASA-TM-X-699; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Heat-transfer rates on the afterbody of the Apollo reentry configuration have been measured in a low-enthalpy wind tunnel at a Mach number of 8. The data have been presented as the ratio of the measured heat-transfer coefficient on the afterbody to the calculated heat-transfer coefficient at the stagnation point at zero angle of attack. This ratio was found to vary from a low of approximately 0.01 to a maximum of about 0.52 as the angle of attack varied from 0 to 55 deg.
Author
Apollo Spacecraft; Hypersonic Speed; Heat Transfer Coefficients; Afterbodies; Heat Transfer; Aerothermodynamics; Hypersonic Reentry; Wind Tunnel Tests; Reentry Effects

19980227768 NASA Langley Research Center, Hampton, VA USA
Flight Performance of a Spin-Stabilized 20-Inch-Diameter Solid-Propellant Spherical Rocket Motor
Levine, Jack, NASA Langley Research Center, USA; Martz, C. William, NASA Langley Research Center, USA; Swain, Robert L., NASA Langley Research Center, USA; Swanson, Andrew G., NASA Langley Research Center, USA; Sep. 1960; 42p; In English
Report No.(s): NASA-TN-D-441; L-596; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A successful flight test of a spin-stabilized 20-inch-diameter solid-propellant rocket motor having a propellant mass fraction of 0.92 has been made. The motor was fired at altitude after being boosted by a three-stage test vehicle. Analysis of the data indicates that a total impulse of 44,243 pound-second with a propellant specific impulse of approximately 185 was achieved over a total action time of about 12 seconds. These results are shown to be in excellent agreement with data from ground static firing tests of these motors. The spherical rocket motor with an 11-pound payload attained a velocity of 15,620 feet per second (m = 16.7) with an incremental velocity increase for the spherical motor stage of 12,120 feet per second.
Author
Solid Propellant Rocket Engines; Flight Characteristics; Flight Tests; Propulsion System Performance

11
CHEMISTRY AND MATERIALS

Includes chemistry and materials (general); composite materials; inorganic and physical chemistry; metallic materials; nonmetallic materials; propellants and fuels; and materials processing.

19980227110 NASA Goddard Space Flight Center, Greenbelt, MD USA

Algorithm for Estimating the Plume Centerline Temperature and Ceiling Jet Temperature in the Presence of a Hot Upper Layer

Davis, William D., National Inst. of Standards and Technology, USA; Notarianni, Kathy A., National Inst. of Standards and Technology, USA; Tapper, Phillip Z., NASA Goddard Space Flight Center, USA; Jun. 1998; 34p; In English

Report No.(s): PB98-146152; NISTIR-6178; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The experiments were designed to provide insight into the behavior of jet fuel fires in aircraft hangars and to study the impact of these fires on the design and operation of a variety of fire protection systems. As a result, the test series included small fires designed to investigate the operation of UV/IR detectors and smoke detectors as well as large fires which were used to investigate the operation of ceiling mounted heat detectors and sprinklers. The impact of the presence or absence of draft curtains was also studied in the 15 m hangar. It is shown that in order to predict the plume centerline temperature within experimental uncertainty, the entrainment of the upper layer gas must be modeled. For large fires, the impact of a changing radiation fraction must also be included in the calculation. The dependence of the radial temperature profile of the ceiling jet as a function of layer development is demonstrated and a ceiling jet temperature algorithm which includes the impact of a growing layer is developed.

DTIC

Plumes; Algorithms; Temperature Profiles; Hangars; Full Scale Tests; Estimating

19980227194 NASA Lewis Research Center, Cleveland, OH USA

Recombination of Hydrogen-Air Combustion Products in an Exhaust Nozzle

Lezberg, Erwin A., NASA Lewis Research Center, USA; Lancashire, Richard B., NASA Lewis Research Center, USA; Aug. 1961; 38p; In English

Report No.(s): NASA-TN-D-1052; E-1246; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Thrust losses due to the inability of dissociated combustion gases to recombine in exhaust nozzles are of primary interest for evaluating the performance of hypersonic ramjets. Some results for the expansion of hydrogen-air combustion products are described. Combustion air was preheated up to 33000 R to simulate high-Mach-number flight conditions. Static-temperature measurements using the line reversal method and wall static pressures were used to indicate the state of the gas during expansion. Results indicated substantial departure from the shifting equilibrium curve beginning slightly downstream of the nozzle throat at stagnation pressures of 1.7 and 3.6 atmospheres. The results are compared with an approximate method for determining a freezing point using an overall rate equation for the oxidation of hydrogen.

Author

Combustion Products; Hydrogen; Stagnation Pressure; Static Pressure; Hypersonics; Ramjet Engines

12
ENGINEERING

Includes engineering (general); communications and radar; electronics and electrical engineering; fluid mechanics and heat transfer; instrumentation and photography; lasers and masers; mechanical engineering; quality assurance and reliability; and structural mechanics.

19980223039 Nagasaki Univ., The Faculty of Engineering, Japan

Characteristics of Fluid Dynamics and Noise in Laminar Flow Fans (Effects of Diameter of Impeller on Characterisitcs)

Kodama, Yoshio, Nagasaki Univ., Japan; Hayashi, Hidechito, Nagasaki Univ., Japan; Tanaka, Kiyohiro, Nagasaki Univ., Japan; Fukui, Tomomi, Nagasaki Univ., Japan; Murahata, Kazuhiro, Nagasaki Univ., Japan; Reports of the Faculty of Engineering, Nagasaki University; Jan. 1993; ISSN 0286-0902; Volume 23, No. 40, pp. 17-23; In Japanese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The effects of six design parameters, the diameter of impeller, the rotational frequency, the gap of two disks, the number of disks, the clearance between casing wall and front shroud, the disk thickness on pressure coefficient were theoretically clarified over a wide range of fan flow rates and the scroll of casing, the diameter of impeller on the noise radiated from fan. The agreement between the predicted and experimental results of the pressure coefficient is satisfactory if the modified equation of velocity ratio

V(uth)/u = 1 and empirical equation of K(m) were used. The experimental results show that the fluid dynamic characteristics were improved and the sound pressure level risen by increasing the diameter of impeller.
Author
Turbofans; Design Analysis; Aerodynamic Noise; Laminar Flow; Flow Characteristics; Impellers; Sound Pressure

19980223040 **Nagasaki Univ., The Faculty of Engineering, Japan**
Characteristics of Fluid Dynamics and Noise in Counter-Rotating Fan
Kodama, Yoshio, Nagasaki Univ., Japan; Hayashi, Hidechito, Nagasaki Univ., Japan; Tanaka, Kiyohiro, Nagasaki Univ., Japan; Yamaguti, Akihiro, Nagasaki Univ., Japan; Reports of the Faculty of Engineering, Nagasaki University; Jan. 1993; ISSN 0286-0902; Volume 23, No. 40, pp. 9-15; In Japanese; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

The effects of asymmetry of the electric motor support, the distance between two impellers on the fluid dynamic characteristics and the fan noise were investigated experimentally with a counter rotating fan. Moreover, the comparison of the fan noise and the fluid dynamic characteristics between the counter rotating fan and a two stage rotor fan was made. It is concluded from these experimental results that the fan noise generated from symmetric support fan is lower 3 to 6 dB than that of asymmetric support fan and the fluid dynamic characteristics of the former is superior to that of the latter. The distance between two impellers is larger, the fan efficiency and the fan noise become lower. The fluid dynamic characteristics of the counter rotating fan with 9-blades is superior to that of the two stage rotor fan, but the noise generated from the former is higher than that of the latter.
Author
Fan Blades; Electric Motors; Counter Rotation; Aerodynamic Noise; Fluid Flow

19980223589 **NASA Langley Research Center, Hampton, VA USA**
Tire-to-Surface Friction-Coefficient Measurements with a C-123B Airplane on Various Runway Surfaces
Sawyer, Richard H., NASA Langley Research Center, USA; Kolnick, Joseph J., NASA Langley Research Center, USA; 1959; 36p; In English
Report No.(s): NASA-TR-R-20; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted to obtain information on the tire-to-surface friction coefficients available in aircraft braking during the landing run. The tests were made with a C-123B airplane on both wet and dry concrete and bituminous pavements and on snow-covered and ice surfaces at speeds from 12 to 115 knots. Measurements were made of the maximum (incipient skidding) friction coefficient, the full-skidding (locked wheel) friction coefficient, and the wheel slip ratio during braking.
Author
Runways; Coefficient of Friction; Pavements; Tires; C-123 Aircraft

19980223601 **NASA Langley Research Center, Hampton, VA USA**
Heat-Transfer and Pressure Measurements on a Flat-Face Cylinder at a Mach Number Range of 2.49 to 4.44
Burbank, Paige B., NASA Langley Research Center, USA; Stallings, Robert L., Jr., NASA Langley Research Center, USA; Aug. 1959; 24p; In English
Report No.(s): NASA-TM-X-19; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Heat-transfer coefficients and pressure distributions were obtained on a 4-inch-diameter flat-face cylinder in the Langley Unitary Plan wind tunnel. The measured stagnation heat-transfer coefficient agrees well with 55 percent of the theoretical value predicted by the modified Sibulkin method for a hemisphere. Pressure measurements indicated the dimensionless velocity gradient parameter r du\ a(sub t) dx, where x=0 at the stagnation point was approximately 0.3 and invariant throughout the Mach number range from 2.49 to 4.44 and the Reynolds number range from 0.77 x 10(exp 6) to 1.46 x 10(exp 6). The heat-transfer coefficients on the cylindrical afterbody could be predicted with reasonable accuracy by flat-plate theory at an angle of attack of 0 deg. At angles of attack the cylindrical afterbody stagnation-line heat transfer could be computed from swept-cylinder theory for large distances back of the nose when the Reynolds number is based on the distance from the flow reattachment points.
Author
Heat Transfer; Pressure Measurement; Mach Number; Aerodynamic Heat Transfer; Flat Plates; Stagnation Point; Aerothermodynamics

19980223696 **NASA Ames Research Center, Moffett Field, CA USA**
A Study of the Simulation of Flow with Free-Stream Mach Number 1 in a Choked Wind Tunnel
Spreiter, John R., NASA Ames Research Center, USA; Smith, Donald W., NASA Ames Research Center, USA; Hyett, B. Jeanne, NASA Ames Research Center, USA; 1960; 40p; In English
Report No.(s): NASA-TR-R-73; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The degree to which experimental results obtained under choking conditions in a wind tunnel with solid walls simulate those associated with an unbounded flow with free-stream Mach number 1 is investigated for the cases of two-dimensional and axisymmetric flows. It is found that a close resemblance does indeed exist in the vicinity of the body, and that the results obtained in this way are generally at least as accurate as those obtained in a transonic wind tunnel with partly open test section. Some of the results indicate, however, that substantial interference effects may be encountered wider certain conditions, both in choked wind tunnels and in transonic wind tunnels, and that reduction of these interference effects to acceptable limits may require the use of models of unusually small size.

Author

Axisymmetric Flow; Transonic Speed; Test Chambers; Transonic Wind Tunnels; Free Flow

19980223610 NASA Lewis Research Center, Cleveland, OH USA

Flow in the Base Region of Axisymmetric and Two-Dimensional Configurations

Beheim, Milton A., NASA Lewis Research Center, USA; 1961; 34p; In English

Report No.(s): NASA-TR-R-77; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A theoretical and experimental investigation has been conducted of the pressure distribution on the surface of either a circular cylinder or a truncated cone located within the base region of another circular cylinder at Mach number 2. A similar analysis of pressure distribution was made for rearward-facing two-dimensional steps, and theoretical results were compared with experimental results of earlier investigations. Effects of base bleed were also studied with the axisymmetric configurations.

Author (revised)

Backward Facing Steps; Circular Cylinders; Supersonic Speed; Pressure Distribution; Boundary Layer Flow; Circular Cones; Aerodynamic Configurations

19980223969 NASA Lewis Research Center, Cleveland, OH USA

Effect of External Boundary Layer on Performance of Axisymmetric Inlet at Mach Numbers of 3.0 and 2.5

Samanich, N. E., NASA Lewis Research Center, USA; Barnett, D. O., NASA Lewis Research Center, USA; Salmi, R. J., NASA Lewis Research Center, USA; Sep. 1959; 18p; In English

Report No.(s): NASA-TM-X-49; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The effect of an external boundary layer on the performance of an axisymmetric external-internal-compression inlet was evaluated at Mach numbers of 3.0 and 2.5 and Reynolds numbers from 2.2 to 0.5 x 10(exp 6) per foot. The inlet was tested at locations up to two-thirds of the way into the 1.7- and 9.0-inch boundary layers generated by a flat plate and the tunnel floor, respectively. The inlet could be readily started at all conditions tested, including those where the boundary layer was separated upstream of the inlet by the various shock systems during the restart cycle. Although the inlet performance decreased with increasing immersion into the boundary layer at both Mach numbers, the inlet was more sensitive to boundary-layer ingestion at the design Mach number of 3.0.

Author

Aircraft Structures; Structural Design; Supersonic Speed; Boundary Layers; Internal Compression Inlets; Supersonic Inlets; Axisymmetric Bodies; Wind Tunnel Tests

19980223975 NASA Langley Research Center, Hampton, VA USA

Elevated-Temperature Tests Under Static and Aerodynamic Conditions on Corrugated-Stiffened Panels

Groen, Joseph M., NASA Langley Research Center, USA; Rosecrans, Richard, NASA Langley Research Center, USA; Sep. 1959; 38p; In English

Report No.(s): NASA-TM-X-34; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Thermal-insulating panels made of 0.005-inch-thick corrugated-stiffened sheets of Inconel X, backed by either bulk or reflective insulation, were tested under static and aerodynamic conditions at elevated temperatures up to 1,8000 F in front of a quartz-tube radiant heater and in a blowdown wind tunnel at a Mach number of 1.4. The tests were performed to provide information on the structural integrity and insulating effectiveness of thermal-insulating panels under the effects of aerodynamic heating. Static radiant-heating tests showed that the bulk insulation protected a load-carrying structure better than did the reflective insulation; however, the bulk insulation was much heavier than the reflective insulation and made the panel assemblies about three times as thick. Three of the four panels tested in the heated supersonic wind tunnel fluttered and failed dynamically. However, one panel demonstrated that flutter can be alleviated considerably with proper edge support. The panels deflected toward the heater (or into the airstream) at a rate which was primarily dependent on the temperature difference through the panel thickness.

Author

Wind Tunnel Tests; Aerodynamic Heating; High Temperature Tests; Insulated Structures; Static Tests; Dynamic Tests; Inconel (Trademark); Thermal Degradation; Supersonic Speed

19980223976 **NASA Ames Research Center, Moffett Field, CA USA**

An Experimental Investigation of Boundary-Layer Control for Drag Reduction of a Swept-Wing Section at Low Speed and High Reynolds Numbers

Gault, Donald E., NASA Ames Research Center, USA; Oct. 1960; 20p; In English

Report No.(s): NASA-TN-D-320; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of laminar boundary-layer control by suction for purposes of drag reduction at low speed and high Reynolds numbers has been conducted in the Ames 12-Foot Pressure Wind Tunnel. The model was a 72.96-inch-chord wing panel, swept back 30 deg., which was installed between end plates to approximate a wing of infinite span. The airfoil section employed was a modified NACA 66-012 in the streamwise direction. Tests were limited to controlling the flow over only the upper surface of the model. Seventeen individually controllable suction chambers were provided below the surface to induce flow through 93 spanwise slots in the surface between the 0.0052- and 0.97-chord stations. Tests were made at angles of attack of 0 deg., +/- 1.0 deg., +/- 1.5 deg., and -2.0 deg. for Reynolds numbers from approximately 1.5 x 10(exp 6) to 4.0 x 10(exp 6) per foot. In general, essentially full-chord laminar flow was obtained for all conditions with small suction quantities. Minimum profile-drag coefficients of about 0.0005 to 0.0006 were obtained for the slotted surface at maximum values of the Reynolds number; these values include the Power required to induce suction as an equivalent drag.

Author

Boundary Layer Control; Aerodynamic Drag; Laminar Boundary Layer; Swept Wings; Wind Tunnel Tests; Suction; Wing Panels; Drag Reduction; End Plates

19980227078 **NASA Ames Research Center, Moffett Field, CA USA**

Forces and Moments on Sphere-Cone Bodies in Newtonian Flow

Dickey, Robert R., NASA Ames Research Center, USA; Dec. 1961; 20p; In English

Report No.(s): NASA-TN-D-1203; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The static longitudinal aerodynamic characteristics of a family of sphere-cone combinations (fineness ratios from 1.0 to 6.0) were computed by means of Newtonian impact theory. The effects of angle of attack, fineness ratio, and center-of-gravity location are shown. The results indicate that, with the center of gravity at or near the center of volume, the sphere-cone combinations are statically stable at trim points that provide low to moderate lift-drag ratios. In general, the lift-drag ratio increased with increasing fineness ratio. As an example, with the center of gravity at the center of volume, the lift-drag ratio at trim was increased from approximately 0.05 to 0.56 by increasing the fineness ratio from 1.2 to 6.0.

Author

Static Aerodynamic Characteristics; Circular Cones; Fineness Ratio; Newton Theory; Longitudinal Stability; Newtonian Fluids; Aerodynamic Forces; Moments; Spheres

19980227098 **NASA Ames Research Center, Moffett Field, CA USA**

Full-Scale Wind-Tunnel Tests of Blowing Boundary-Layer Control Applied to a Helicopter Rotor

McCloud, John L., III, NASA Ames Research Center, USA; Hall, Leo P., NASA Ames Research Center, USA; Brady, James A., NASA Ames Research Center, USA; Sep. 1960; 36p; In English

Report No.(s): NASA-TN-D-335; A-380; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A full-scale wind-tunnel test was conducted of two boundary-layer-control applications to a 44-foot diameter helicopter rotor. Blowing from a nozzle near the leading edge of the blades delayed retreating blade stall. Results also indicated that delay of retreating blade stall could be obtained by cyclic blowing with a lower flow rate than that required for continuous blowing. It was found that blowing applied through a nozzle at mid-chord had no effect on retreating blade stall.

Author

Boundary Layer Control; Full Scale Tests; Wind Tunnel Tests; Blowing; Leading Edges; Rotary Wings

19980227113 **NASA Lewis Research Center, Cleveland, OH USA**

Reduction of Jet Penetration in a Cross-Flow by Using Tabs

Zaman, K. B. M. Q., NASA Lewis Research Center, USA; 1998; 8p; In English; 34th; Propulsion, 13-15 Jul. 1998, Cleveland,

OH, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA
Report No.(s): AIAA Paper 98-3276; No Copyright; Avail: Issuing Activity, Hardcopy, Microfiche

A tab placed suitably on a nozzle that produces a jet in a cross-flow can reduce the penetration of the jet. This effect, achieved when the tab is placed on the windward side of the nozzle relative to the cross flow, may be of interest in film cooling applications. Wind tunnel experiments are carried out, in the momentum ratio (J) range of 10-90, to investigate the tab geometry that would maximize this effect. The preliminary results show that a 'delta tab' having a base width approximately fifty percent of the nozzle diameter may be considered optimum. With a given tab size, the effect is more pronounced at higher J. Reduction in jet penetration by as much as 40% is observed. Comparable reduction in jet penetration is also obtained when a triangular shaped tab is placed flush with the tunnel wall or with its apex tilted down into the jet nozzle (the 'delta tab' being the configuration in which the apex is tilted up). However, the delta tab involves the least flow blockage and pressure loss. Relative to the baseline case, the lateral spreading of the jet is found to be more with the delta tab but less with other orientations of the tab.

Author

Tabs (Control Surfaces); Cross Flow; Flow Characteristics; Flow Geometry; Aerodynamic Characteristics; Nozzle Flow

19980227181 **NASA Langley Research Center, Hampton, VA USA**

An Experimental Investigation and Correlation of the Heat Reduction to Nonporous Surfaces Behind a Porous Leading Edge Through Which Coolant is Ejected

Witte, William G., NASA Langley Research Center, USA; Rashis, Bernard, NASA Langley Research Center, USA; Mar. 1960; 34p; In English
Report No.(s): NASA-TM-X-235; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A configuration of a wing segment having constant chord thickness, 0 deg. sweep, a porous steel semicircular leading edge, and solid Inconel surfaces was tested in a Mach number 2.0 ethlyene-heated high-temperature air jet. Measurements were made of the wing surface temperatures at chordwise stations for several rates of helium flow through the porous leading edge. The investigation was conducted at stagnation temperatures ranging from 500 F to 2,400 F, at Reynolds numbers per foot ranging from 0.3 x 10(exp 7) to 1.2 x 10(exp 7), and at angles of attack of 0, +/- 5, and +/- 15 deg. The results indicated that the reduction of wing surface temperatures with respect to their values for no coolant flow, depended on the helium coolant flow rates and the distance behind the area of injection. The results were correlated in terms of the wall cooling parameter and the coolant flow-rate parameter, where the nondimensional flow rate was referenced to the cooled area up to the downstream position. For the same coolant flow rate, lower surface temperatures are achieved with a porous-wall cooling system. However, since flow-rate requirements decrease with increasing allowable surface temperatures, the higher allowable wall temperatures,of the solid wall as compared to the structurally weaker porous wall- sharply reduce the flow-rate requirements of a downstream cooling system. Thus, for certain flight conditions it is possible to compensate for the lower efficiency of the downstream or solid-wall cooling system. For example, a downstream cooling system using solid walls that must be maintained at 1,800 F would require less coolant for Mach numbers up to 5.5 than would a porous-wall cooling system for which the walls must be maintained at temperatures less than or equal to 9000 F.

Author

Leading Edges; Porous Walls; Cooling Systems; Coolants; Air Jets; Solid Surfaces; Wall Temperature; Wings; Cooling

19980227185 **NASA Ames Research Center, Moffett Field, CA USA**

Effects of Sweep Angle on the Boundary-Layer Stability Characteristics of an Untapered Wing at Low Speeds

Boltz, Frederick W., NASA Ames Research Center, USA; Kenyon, George C., NASA Ames Research Center, USA; Allen, Clyde Q., NASA Ames Research Center, USA; Oct. 1960; 80p; In English
Report No.(s): NASA-TN-D-338; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An investigation was conducted in the Ames 12-Foot Low-Turbulence Pressure Tunnel to determine the effects of sweep on the boundary-layer stability characteristics of an untapered variable-sweep wing having an NACA 64(2)A015 section normal to the leading edge. Pressure distribution and transition were measured on the wing at low speeds at sweep angles of 0, 10, 20, 30, 40, and 50 deg. and at angles of attack from -3 to 3 deg. The investigation also included flow-visualization studies on the surface at sweep angles from 0 to 50 deg. and total pressure surveys in the boundary layer at a sweep angle of 30 deg. for angles of attack from -12 to 0 deg. It was found that sweep caused premature transition on the wing under certain conditions. This effect resulted from the formation of vortices in the boundary layer when a critical combination of sweep angle, pressure gradient, and stream Reynolds number was attained. A useful parameter in indicating the combined effect of these flow variables on vortex formation and on beginning transition is the crossflow Reynolds number. The critical values of crossflow Reynolds number for vortex formation found in this investigation range from about 135 to 190 and are in good agreement with those reported in previous investigations. The values of crossflow Reynolds number for beginning transitions were found to be between 190 and 260. For each condition (i.e., development of vortices and initiation of transition at a given location) the lower values in the specified ranges were obtained with a light coating of flow-visualization material on the surface. A method is presented for the rapid computation of crossflow Reynolds number on any swept surface for which the pressure distribution is known. From calculations based on this method, it was found that the maximum values of crossflow Reynolds number are attained under conditions of a strong pressure gradient and at a sweep angle of about 50 deg. Due to the primary dependence on pressure gradient, effects of sweep in causing premature transition are generally first encountered on the lower surfaces of wings operating at positive angles of attack.

Author

Variable Sweep Wings; Sweep Angle; Boundary Layer Stability; Wind Tunnel Tests; Flow Visualization; Cross Flow; Aerodynamic Stability

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19980227274 **NASA Langley Research Center, Hampton, VA USA**

Laminar Heat-Transfer and Pressure Measurements at a Mach Number of 6 on Sharp and Blunt 15 deg Half-Angle Cones at Angles of Attack Up to 90 deg

Conti, Raul J., NASA Langley Research Center, USA; Oct. 1961; 34p; In English

Report No.(s): NASA-TN-D-962; L-1624; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Two circular conical configurations having 15 deg half-angles were tested in laminar boundary layer at a Mach number of 6 and angles of attack up to 90 deg. One cone had a sharp nose and a fineness ratio of 1.87 and the other had a spherically blunted nose with a bluntness ratio of 0.1428 and a fineness ratio of 1.66. Pressure measurements and schlieren pictures of the flow showed that near-conical flow existed up to an angle of attack of approximately 60 deg. At angles of attack above 70 deg high-pressure areas were present near the base and the bow shock wave was considerably curved. Comparison of the results with simply applied theories showed that on the stagnation line pressures may be predicted by Newtonian theory, and heat transfer by local yawed-cylinder theory based on the yaw angle of the windward generator and the local radius of the cone. Base effects increased the heat transfer in a region extending forward approximately 15 to 30 percent of the windward generator. Circumferential pressure distributions were higher than the corresponding Newtonian distribution and a better prediction was obtained by modifying the theory to match the pressure at 90 deg from the windward generator to that on the surface of the cone at an angle of attack of 0 deg. Circumferential heat-transfer distributions were predicted satisfactorily up to about 60 deg from the stagnation line by using Lees' heat-flux distribution based on the Newtonian pressure. The effects of nose bluntness at large angles of attack were very small in the region beyond two nose radii from the point of tangency.

Author

Laminar Boundary Layer; Heat Transfer; Pressure Measurement; Half Cones; Conical Flow; Angle of Attack; Pressure Distribution; Stagnation Pressure

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19980227276 **NASA Langley Research Center, Hampton, VA USA**

Pressure Loads Produced on a Flat-Plate Wing by Rocket Jets Exhausting in a Spanwise Direction Below the Wing and Perpendicular to a Free-Stream Flow of Mach Number 2.0

Falanga, Ralph A., NASA Langley Research Center, USA; Janos, Joseph J., NASA Langley Research Center, USA; May 1961; 46p; In English

Report No.(s): NASA-TN-D-893; L-1614; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation at a Reynolds number per foot of 14.4 x 10(exp 6) was made to determine the pressure loads produced on a flat-plate wing by rocket jets exhausting in a spanwise direction beneath the wing and perpendicular to a free-stream flow of Mach number 2.0. The ranges of the variables involved were (1) nozzle types - one sonic (jet Mach number of 1.00), two supersonic (jet Mach numbers of 1.74 and 3.04), and one two-dimensional supersonic (jet Mach number of 1.71); (2) vertical nozzle positions beneath the wing of 4, 8 and 12 nozzle-throat diameters; and (3) ratios of rocket-chamber total pressure to free- stream static pressure from 0 to 130. The incremental normal force due to jet interference on the wing varied from one to two times the rocket thrust and generally decreased as the pressure ratio increased. The chordwise coordinate of the incremental-normal-force center of pressure remained upstream of the nozzle center line for the nozzle positions and pressure ratios of the investigation. The chordwise coordinate approached zero as the jet vertical distance beneath the wing increased. In the spanwise direction there was little change due to varying rocket-jet position and pressure ratio. Some boundary-layer flow separation on the wing was observed for the rocket jets close to the wing and at the higher pressure ratios. The magnitude of the chordwise and spanwise pressure distributions due to jet interference was greatest for rocket jets close to the wing and decreased as the jet was displaced farther from the wing. The design procedure for the rockets used is given in the appendix.

Author

Aerodynamic Interference; Boundary Layer Separation; Center of Pressure; Pressure Distribution; Pressure Ratio; Supersonic Jet Flow; Rocket Thrust; Boundary Layer Flow

19980227304 NASA Lewis Research Center, Cleveland, OH USA
Turbulence Studies of a Rectangular Slotted Noise-Suppressor Nozzle
Laurence, James C., NASA Lewis Research Center, USA; Sep.1960; 90p; In English
Report No.(s): NASA-TN-D-294; E-384; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The problem of noise suppression of turbojet engines has shown a need for turbulence data within the flow field of various types of nozzles used in ad hoc investigations of the sound power. The result of turbulence studies in a nozzle configuration of four parallel rectangular slots is presented in this report with special attention to the effect of the spacing of the nozzles on the intensity of turbulence, scale of turbulence, spectrum of turbulence, and the mean stream velocity. Taylor's hypothesis, which describes the convection of the turbulence eddies, was tested and found correct within experimental error and certain experimental and theoretical limitations. The convection of the pressure patterns was also investigated, and the value of the convection velocity was found to be about 0.43 times the central core velocity of the jets. The effect of the spacing-to-width ratio of the nozzles upon the turbulence intensity, the scale of turbulence, and the spectral distribution of the noise was found in general to produce a maximum change for spacing-to-width ratios of 1.5 to 2.0. These changes may be the cause of the reduction in sound power reported for similar full-scale nozzles and test conditions under actual (static) engine operation. A noise reduction parameter is defined from Lighthill's theory which gives qualitative agreement with experiments which show the noise reduction is greatest for spacing-to-width ratios of 1.5 to 2.0.

Author

Noise Reduction; Turbulence; Nozzles; Turbojet Engines; Flow Velocity; Full Scale Tests; Flow Distribution

19980227317 Army Tank-Automotive Research and Development Command, Warren, MI USA
Lab Test of MIPS Turbodyne II Precleaner with Seavenge Blower Motor, *Jan. - May. 1997*
Richard, Michael; McDuffee, Michael; Sierpien, Larry; Margrif, Frank; Jul. 1998; 131p; In English
Report No.(s): AD-A350739; TARDEC-TR-13752; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

An engine induction air precleaner system designed for the MIPS was lab tested at both TARDEC and SWRI to measure pressure drop, efficiency and particle size determination. The Turbodyne II Precleaner is a two-stage precleaner installed up-stream of turbocharged diesel engines. A self-cleaning rotating barrier filter is the second component of the Turbodyne II self-cleaning air filter (SCAF) system and is installed after the turbocharger. Two previous tests of the Turbodyne II SCAF were conducted: (1) Reference Appendix A, report page and abstract and (2) Reference Appendix B, report page and abstract Test results showed in general: (1) Turbocharger degradation does not occur when exposed to precleaned air and (2) some minor difficulty occurred in achieving normal efficiency requirements and/or pressure drop limits across SCAF barrier filter for up to 200 hours. TARDEC and SwRI pressure drop lab tests were in agreement reaching a maximum of 11.2 to 11.4 inches of water at rated flow of 2600 cfm. Likewise efficiency testing at TARDEC and SwRI conducted on PTI coarse test dust was nearly in agreement with TARDEC obtaining an average overall efficiency of 98.15% compared to the slightly higher average overall efficiency of 98.624% obtained by SwRI. Turbodyne II precleaner particle size determination tests conducted by SwRI (See Appendix G, report) showed for three dust concentrations (zero visibility, half zero visibility and quarter zero visibility) separation efficiency at low concentration levels becomes more sensitive to airflow. For the three dust concentrations tested, test results showed the precleaner had an effective cut size ranging from about 3 to 6.5 microns depending on concentration and airflow rate. The cut size is the particle size where the probability of collection is 50%.

DTIC

Blowers; Cleaning; Air Filters; Superchargers

19980227345 Physical Research, Inc., Kirkland, WA USA
Development of an Improved Magneto-Optic/Eddy-Current Imager *Final Report*
Thome, David K., Physical Research, Inc., USA; Apr. 1998; 54p; In English
Contract(s)/Grant(s): DTRS57-95-C-00086
Report No.(s): AD-A350709; DOT/FAA/AR-97/37; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Magneto-optic/eddy-current imaging technology has been developed and approved for inspection of cracks in aging aircraft. This relatively new nondestructive test method gives the inspector the ability to quickly generate real-time eddy-current images of large surface areas. An earlier Phase I Small Business Innovative Research (SBIR) program demonstrated the ability to generate improved, complete, real-time magneto-optic/eddy-current images of subsurface corrosion and cracking. Multidirectional eddy-current excitation, enhanced low-frequency operation, improved electromagnetic shielding, image processing, and sensor improvement were all demonstrated or evaluated. Favorable results from Phase I led to this Phase II SBIR program. The Phase II research has resulted in the development of a next generation prototype magneto-optic imager (MOI) with multidirectional eddy-current excitation, remotely programmable system settings, on-screen display of system setup information, and improved

shielding to, enhanced images. Some of these new features have already been successfully incorporated into an improved imager, the MOI 303, which is now commercially available.
DTIC
Eddy Currents; Image Processing; Imaging Techniques; Magneto-Optics; Corrosion; Nondestructive Tests; Aircraft Maintenance; Images

19980227349 NASA Lewis Research Center, Cleveland, OH USA
Estimate of Shock Standoff Distance Ahead of a General Stagnation Point
Reshotko, Eli, NASA Lewis Research Center, USA; Aug. 1961; 18p; In English
Report No.(s): NASA-TN-D-1050; E-1278; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The shock standoff distance ahead of a general rounded stagnation point has been estimated under the assumption of a constant-density-shock layer. It is found that, with the exception of almost-two-dimensional bodies with very strong shock waves, the present theoretical calculations and the experimental data of Zakkay and Visich for toroids are well represented by the relation Delta-3D/R(s) = ((Delta-ax sym)/(R(s))/(2/(K+1))) where Delta is the shock standoff distance, R(s),x is the smaller principal shock radius, and K is the ratio of the smaller to the larger of the principal shock radii.
Author
Shock Waves; Two Dimensional Bodies; Blunt Bodies; Stagnation Point; Inviscid Flow; Aerodynamic Configurations; Symmetrical Bodies

19980227352 NASA Langley Research Center, Hampton, VA USA
A Method of Solution with Tabulated Results for the Attached Oblique Shock-Wave System for Surfaces at Various Angles of Attack, Sweep, and Dihedral in an Equilibrium Real Gas Including the Atmosphere
Trimpi, Robert L., NASA Langley Research Center, USA; Jones, Robert A., NASA Langley Research Center, USA; 1960; 142p; In English
Report No.(s): NASA-TR-R-63; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

A new method of solution is derived from basic physical considerations. Results are tabulated for the following ranges: angle of attack, 0 deg to 65 deg; angle of sweep, 0 deg to 75 deg; angle of dihedral, 0 deg to 30 deg; Mach number, 3 to 3O;and "effective specific-heat ratio"parameter, 1.10 to 1.67. Both the method and tabulated solutions are easily adaptable to flight in any gas or in the atmosphere of any planet. An illustrative example is presented based on the 1956 ARDC model atmosphere.
Author
Angle of Attack; Real Gases; Atmospheric Models; Dihedral Angle

19980227400 NASA Langley Research Center, Hampton, VA USA
Configuration Factors for Exchange of Radiant Energy Between Axisymmetrical Sections of Cylinders, Cones, and Hemispheres and Their Bases
Buschman, Albert J., Jr., NASA Langley Research Center, USA; Pittman, Claud M., NASA Langley Research Center, USA; Oct. 1961; 48p; In English
Report No.(s): NASA-TN-D-944; L-992; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Radiation-interchange configuration factors are derived for axisymmetrical sections of cylinders, cones, and hemispheres radiating internally to annular and circular sections of their bases and to other axisymmetrical sections. The general procedure of obtaining configuration factors is outlined and the results are presented in the form of equations, tables, and figures.
Author
Heat Transfer; Heat Transfer Coefficients; Cones; Aerodynamic Configurations; Radiant Heating; Cylindrical Bodies

19980227408 NASA Langley Research Center, Hampton, VA USA
Local Aerodynamic Heat Transfer and Boundary-Layer Transition on Roughened Sphere-Ellipsoid Bodies at Mach Number 3.0
Deveikis, William D., NASA Langley Research Center, USA; Walker, Robert W., NASA Langley Research Center, USA; Aug. 1961; 26p; In English
Report No.(s): NASA-TN-D-907; L-1393; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation was made to determine heat-transfer distributions on three steel sphere-ellipsoid bodies with surface roughnesses of 5, 100, and 200 microinches. Tests were conducted in the Langley 9- by 6-foot thermal structures tunnel at a Mach number of 3.0, free-stream Reynolds numbers (based on model spherical diameter) of 4.25 x 10(exp 6) and 2.76 x 10(exp 6), and at a stagnation temperature of 650 F. Pressure distributions were obtained also on a fourth model. The results indicated

that the combination of surface roughness and boundary-layer cooling tended to promote early transition and nullify the advantages attributable to the blunt shape of the model for reducing local temperatures. Good correlation between experimental heating rates and those calculated from laminar theory was achieved up to the start of boundary-layer transition. The correlation also was good with the values predicted by turbulent theory for surface stations downstream from the 45 deg. station.
Author
Aerodynamic Heat Transfer; Boundary Layer Transition; Spheres; Supersonic Speed; Wind Tunnel Tests; Surface Roughness; Free Flow; Cooling

19980227751 NASA Langley Research Center, Hampton, VA USA
Free-Flight Investigation of Heat Transfer to an Unswept Cylinder Subjected to an Incident Shock and Flow Interference from an Upstream Body at Mach Numbers up to 5.50
Carter, Howard S., NASA Langley Research Center, USA; Carr, Robert E., NASA Langley Research Center, USA; Oct. 1961; 34p; In English
Report No.(s): NASA-TN-D-988; L-879; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Heat-transfer rates have been measured in free flight along the stagnation line of an unswept cylinder mounted transversely on an axial cylinder so that the shock wave from the hemispherical nose of the axial cylinder intersected the bow shock of the unswept transverse cylinder. Data were obtained at Mach numbers from 2.53 to 5.50 and at Reynolds numbers based on the transverse cylinder diameter from 1.00 x 10(exp 6) to 1.87 x 10(exp 6). Shadowgraph pictures made in a wind tunnel showed that the flow field was influenced by boundary-layer separation on the axial cylinder and by end effects on the transverse cylinder as well as by the intersecting shocks. Under these conditions, the measured heat-transfer rates had inconsistent variations both in magnitude and distribution which precluded separating the effects of these disturbances. The general magnitude of the measured heating rates at Mach numbers up to 3 was from 0.1 to 0.5 of the theoretical laminar heating rates along the stagnation line for an infinite unswept cylinder in undisturbed flow. At Mach numbers above 4 the measured heating rates were from 1.5 to 2 times the theoretical rates.
Author
Free Flight; Wind Tunnel Tests; Supersonic Speed; Shock Waves; Heat Transfer; Aerodynamic Heating; Bow Waves; Cylindrical Bodies; Unswept Wings

19980227735 NASA Langley Research Center, Hampton, VA USA
Preliminary Investigation of an Underwater Ramjet Powered by Compressed Air
Mottard, Elmo J., NASA Langley Research Center, USA; Shoemaker, Charles J., NASA Langley Research Center, USA; Dec. 1961; 36p; In English
Report No.(s): NASA-TN-D-991; L-1249; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Part I contains the results of a preliminary experimental investigation of a particular design of an underwater ramjet or hydroduct powered by compressed air. The hydroduct is a propulsion device in which the energy of an expanding gas imparts additional momentum to a stream of water through mixing. The hydroduct model had a fineness ratio of 5.9, a maximum diameter of 3.2 inches, and a ratio of inlet area to frontal area of 0.32. The model was towed at a depth of 1 inch at forward speeds between 20 and 60 feet per second for airflow rates from 0.1 to 0.3 pound per second. Longitudinal force and pressures at the inlet and in the mixing chamber were determined. The hydroduct produced a positive thrust-minus-drag force at every test speed. The force and pressure coefficients were functions primarily of the ratio of weight airflow to free-stream velocity. The maximum propulsive efficiency based on the net internal thrust and an isothermal expansion of the air was approximately 53 percent at a thrust coefficient of 0.10. The performance of the test model may have been influenced by choking of the exit flow. Part II is a theoretical development of an underwater ramjet using air as "fuel." The basic assumption of the theoretical analysis is that a mixture of water and air can be treated as a compressible gas. More information on the properties of air-water mixtures is required to confirm this assumption or to suggest another approach. A method is suggested from which a more complete theoretical development, with the effects of choking included, may be obtained. An exploratory computation, in which this suggested method was used, indicated that the effect of choked flow on the thrust coefficient was minor.
Author
Ramjet Engines; Propulsion; Compressed Air; Free Flow; Propulsive Efficiency; Thrust

19980227763 NASA Langley Research Center, Hampton, VA USA
Cross-Sectional Deformations of Monocoque Beams and Their Effects on the Natural Vibration Frequencies
Thomson, Robert G., NASA Langley Research Center, USA; Kruszewski, Edwin T., NASA Langley Research Center, USA; Dec. 1961; 50p; In English

Report No.(s): NASA-TN-D-987; L-1444; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The variational principle, differential equations, and boundary conditions governing the cross-sectional distortions due to inertia loading of a two-dimensional model of a thin monocoque wing are shown. A theoretical analysis of this simplified model is made in order to determine the nature of the coupling between the cross-sectional modes and the spanwise deformation modes. General solutions are obtained in finite-difference form for arbitrary cross sections and an exact solution is presented for a parabolic-arc cross section of constant cover thickness. The application of these results in evaluating the coupled frequencies of the actual structure is discussed. Frequencies evaluated for a parabolic-arc monocoque beam show good agreement with experimental values.

Author

Monocoque Structures; Aircraft Structures; Dynamic Structural Analysis; Finite Difference Theory; Structural Vibration; Deformation

19980227795 NASA Langley Research Center, Hampton, VA USA

Experimental Investigation at Mach Number 3.0 of the Effects of Thermal Stress and Buckling on the Flutter of Four-Bay Aluminum Alloy Panels with Length-Width Ratios of 10

Dixon, Sidney C., NASA Langley Research Center, USA; Griffith, George E., NASA Langley Research Center, USA; Bohon, Herman L., NASA Langley Research Center, USA; Oct. 1961; 30p; In English

Report No.(s): NASA-TN-D-921; L-1265; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Skin-stiffener aluminum alloy panels consisting of four bays, each bay having a length-width ratio of 10, were tested at a Mach number of 3.0 at dynamic pressures ranging from 1,500 psf to 5,000 psf and at stagnation temperatures from 300 F to 655 F. The panels were restrained by the supporting structure in such a manner that partial thermal expansion of the skins could occur in both the longitudinal and lateral directions. A boundary faired through the experimental flutter points consisted of a flat-panel portion, a buckled-panel portion, and a transition point at the intersection of the two boundaries. In the region where a panel must be flat when flutter occurs, an increase in panel skin temperature (or midplane compressive stress) makes the panel more susceptible to flutter. In the region where a panel must be buckled when flutter occurs, the flutter trend is reversed. This reversal in trend is attributed to the panel postbuckling behavior.

Author

Aluminum Alloys; Panels; Supersonic Speed; Buckling; Thermal Stresses; Panel Flutter; Vibrational Stress; Aeroelasticity

14
LIFE SCIENCES

Includes life sciences (general); aerospace medicine; behavioral sciences; man/system technology and life support; and space biology.

19980223621 NASA Ames Research Center, Moffett Field, CA USA

Centrifuge Study of Pilot Tolerance to Acceleration and the Effects of Acceleration on Pilot Performance

Creer, Brent Y., NASA Ames Research Center, USA; Smedal, Harald A., NASA Ames Research Center, USA; Wingrove, Rodney C., NASA Ames Research Center, USA; Nov. 1960; 38p; In English

Report No.(s): NASA-TN-D-337; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A research program the general objective of which was to measure the effects of various sustained accelerations on the control performance of pilots, was carried out on the Aviation Medical Acceleration Laboratory centrifuge, U.S. Naval Air Development Center, Johnsville, PA. The experimental setup consisted of a flight simulator with the centrifuge in the control loop. The pilot performed his control tasks while being subjected to acceleration fields such as might be encountered by a forward-facing pilot flying an atmosphere entry vehicle. The study was divided into three phases. In one phase of the program, the pilots were subjected to a variety of sustained linear acceleration forces while controlling vehicles with several different sets of longitudinal dynamics. Here, a randomly moving target was displayed to the pilot on a cathode-ray tube. For each combination of acceleration field and vehicle dynamics, pilot tracking accuracy was measured and pilot opinion of the stability and control characteristics was recorded. Thus, information was obtained on the combined effects of complexity of control task and magnitude and direction of acceleration forces on pilot performance. These tests showed that the pilot's tracking performance deteriorated markedly at accelerations greater than about 4g when controlling a lightly damped vehicle. The tentative conclusion was also reached that regardless of the airframe dynamics involved, the pilot feels that in order to have the same level of control over the vehicle, an increase in the vehicle dynamic stability was required with increases in the magnitudes of the acceleration impressed upon the pilot. In another phase, boundaries of human tolerance of acceleration were established for acceleration fields such as might be encountered by a pilot flying an orbital vehicle. A special pilot restraint system was developed to increase human tolerance to longitudinal decelerations.

The results of the tests showed that human tolerance of longitudinal deceleration forces was considerably improved through use of the special restraint system.
Author
Pilot Performance; Human Tolerances; Flight Simulators; Deceleration; Dynamic Stability; Atmospheric Entry; Centrifuges

19980223933 **Naval Aerospace Medical Research Lab., Pensacola, FL USA**
Calculating A Helicopter Pilot's Instrument Scan Pattern from Discrete, 60-Hz Measures of the Line-of-Sight: The Evaluation of an Algorithm
Jun. 17, 1998; 33p; In English
Report No.(s): AD-A350657; NAMRL-1403; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

In order obtain data to develop and evaluate theories relating instrument scanning to flight performance we recorded the line of sight (LOS) of student naval helicopter pilots as they flew prescribed maneuvers in a motion-based, high fidelity, instrument training simulator. These LOS data were discrete, 60 Hz samples of eye pointing. For some types of analysis it is helpful to think of a scan pattern as a sequence of fixations and to use an averaging algorithm to transform the 60 Hz data into such a sequence, a scan path. An appropriate algorithm was identified, developed and evaluated. As part of this evaluation, we developed a String Similarity measure, SS, a measure of the similarity between two scan paths. The evaluation of the algorithm, consisting of observing the algorithm's output as a function of the algorithm's parameter values, showed that the algorithm behaved in a sensible fashion, logically consistent with the input data. This increased our confidence in our implementation of the fixation algorithm. The SS metric proved to be an informative, useful tool that may have addition uses in the analysis scanning behavior and flight performance.
DTIC
Helicopters; Scanners; Flight Instruments; Line of Sight; Algorithms; Aircraft Pilots

19980223984 **Washington Univ., Seattle, WA USA**
The Adaptive Effects of Virtual Interfaces: Vestibulo-Ocular Reflex and Simulator Sickness
Draper, Mark H., Washington Univ., USA; Aug. 07, 1998; 345p; In English
Report No.(s): AD-A350767; AFIT-98-021D; No Copyright; Avail: CASI; A15, Hardcopy; A03, Microfiche

Current virtual interfaces imperfectly simulate the motion dynamics of the real world. Conflicting visual and vestibular cues of self-motion are believed to result in vestibulo-ocular reflex (VOR) adaptations and simulator sickness, which raises health and safety issues surrounding virtual environment (VE) exposure. Four experiments were conducted to examine the effects of conflicting visual-vestibular cues through employment of typically occurring virtual interface scenarios. Subjects were exposed for 30 minutes to a head-coupled virtual interface, completing visual search tasks using active, unrestricted head movement rotations.
DTIC
Virtual Reality; Visual Perception; Motion Sickness; Flight Simulators; Reflexes

19980227270 **Army Aeromedical Research Lab., Fort Rucker, AL USA**
Effects of Head-Supported Devices on Female Aviators during Simulated Helicopter Missions *Annual Report*
Alem, Nabih, Army Aeromedical Research Lab., USA; May 1998; 122p; In English
Report No.(s): AD-A350472; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

This report describes the work completed during the first project year of this research study. The objective of the study is to identify safe weight and location limits of head-supported devices worn by female aviators during simulated helicopter rides. The working hypothesis is that female pilots will tolerate some range of HSD weight moments beyond which their biomechanical and performance responses will deteriorate. The report contains a review of relevant studies followed by detailed description of the experimental and analytical procedures.
DTIC
Aircraft Pilots; Females; Helicopters; Helmets

19980227326 **Air Force Research Lab., Human Effectiveness Directorate, Wright-Patterson AFB, OH USA**
Building the LeM2*R3 Model of Pilot Trust and Dynamic Workload Allocation: A Transition of Theory and Empirical Observations to Cockpit Demonstration *Final Report, Jan. 1994 - Oct. 1997*
Raeth, Peter G.; Reising, John M.; Feb. 1998; 102p; In English
Contract(s)/Grant(s): Proj-2403
Report No.(s): AD-A350481; AFRL-HE-WP-TR-1998-0046; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

For pilots to accept active decision aids during complex flight scenarios, it is essential that the automation work is in synergy with aircrew. To accomplish this, the automation must go well beyond menu and macro selections, where the pilot must explicitly tell the automation what to do and when to do it. It must also transcend "mother may I" approaches, where the automation asks for permission to proceed. to these traditional barriers, the automation needs a sense of how the pilot will react in a given situation and, based on that reaction, how much of the workload could be allocated to the automation at any given time. For this purpose, the authors reviewed the literature on human factors and dynamic function allocation. This literature provided a wealth of information on this topic. Based on the current state of the art in this topic area, the authors developed and tested a dynamic model of pilot trust and workload allocation. This "full degrees of freedom" model transitions human factors theory, as it exists today, into an engineering application. The resulting model can be combined with other information obtained from static and continuous processes to divide the workload and minimize cognitive overload.
DTIC
Cockpits; Human Factors Engineering; Decision Support Systems; Artificial Intelligence; Decision Making; Flight Simulation

15
MATHEMATICAL AND COMPUTER SCIENCES

Includes mathematical and computer sciences (general); computer operations and hardware; computer programming and software; computer systems; cybernetics; numerical analysis; statistics and probability; systems analysis; and theoretical mathematics.

19980227167 Army Command and General Staff Coll., Fort Leavenworth, KS USA
F-16 Peacetime Training for Combat Operations
Roosa, John D., Army Command and General Staff Coll., USA; Jun. 05, 1998; 98p; In English
Report No.(s): AD-A350054; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

This study investigates the relationship between peacetime F-16 training and expected combat operations. The F-16 is the primary interdiction platform in the USA Air Force. F-16 pilots fly peacetime training sorties to maintain proficiency, develop tactics and complete evaluations. The training activities accomplished on these missions are designed to prepare the pilot for successful combat employment. A training program ensures each pilot completes the necessary amount of sorties and events to achieve combat ready status. This study analyzes the components of the peacetime training program and their overall applicability for future conflict. The study encompasses the entire training program from higher headquarters directives down to specific flying sorties.
DTIC
Peacetime; Flight Training; F-16 Aircraft; Education

19980227173 Army Command and General Staff Coll., Fort Leavenworth, KS USA
Premobilization Proficiency of USA Army Reserve Attack Helicopter Battalions
Gruenwald, David L., Army Command and General Staff Coll., USA; Jun. 05, 1998; 85p; In English
Report No.(s): AD-A349995; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Attack helicopter battalions are combat maneuver units that conduct supporting attacks which aid, protect, and compliment other maneuver forces by destroying massed enemy mechanized forces and other enemy forces with aerial firepower, mobility, and shock effect. They are employed as a battalion in order to provide the commander with this highly mobile and lethal destruction capability. The fundamentals of attack helicopter operations do not change by component. Reserve Component attack helicopter battalions are expected to perform attack helicopter operations to the same level of proficiency or standard as the Active Component. Currently, there is conflicting guidance published by Forces Command as to what level of proficiency aviation units in the Reserve component should train to in premobilization in order to prepare for their wartime mission. This study examines the ability of USA Army Reserve (USAR) attack helicopter units to maintain proficiency at the battalion level in a premobilization environment. It focuses on the resources available to Reserve units and the training requirements placed on a unit. It concludes with an analysis of a USAR attack helicopter unit's ability to execute all training requirements in the time available to them each training year. It offers recommendations on possible alternative training strategies and provides suggestions for further research.
DTIC
Helicopters; Armed Forces (USA); Attack Aircraft; Education

16
PHYSICS

Includes physics (general); acoustics; atomic and molecular physics; nuclear and high-energy; optics; plasma physics; solid-state physics; and thermodynamics and statistical physics.

19980223604 **NASA Lewis Research Center, Cleveland, OH USA**

Similarity of Near Noise Fields of Subsonic Jets

Howes, Walton L., NASA Lewis Research Center, USA; 1961; 56p; In English

Report No.(s): NASA-TR-R-94; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Similarity relations for frequency pass band, as well as overall, time average pressure fluctuations outside a jet are derived and tested using experimental data. Similarity of the pressure fields was found for different jet velocities. Nozzle contour dissimilarity and differing jet temperatures were found to limit seriously the application of the similarity relations, especially near the jet nozzle.

Author

Subsonic Flow; Air Jets; Engine Noise; Near Fields; Pressure Oscillations; Aerodynamic Noise; Pressure Gradients; Sound Pressure

18
SPACE SCIENCES

Includes space sciences (general); astronomy; astrophysics; lunar and planetary exploration; solar physics; and space radiation.

19980223611 **NASA Marshall Space Flight Center, Huntsville, AL USA**

Aerodynamic Analysis of Tektites and Their Parent Bodies

Adams, E. W., NASA Marshall Space Flight Center, USA; Huffaker, R. M., NASA Marshall Space Flight Center, USA; 1962; 48p; In English

Report No.(s): NASA-TR-R-149; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Experiment and analysis indicate that the button-type australites were derived from glassy spheres which entered or re-entered the atmosphere as cold solid bodies; in case of average-size specimens, the entry direction was nearly horizontal and the entry speed between 6.5 and 11.2 km/sec. Terrestrial origin of such spheres is impossible because of extremely high deceleration rates at low altitudes. The limited extension of the strewn fields rules out extraterrestrial origin of clusters of such spheres because of stability considerations for clusters in space. However, tektites may have been released as liquid droplets from glassy parent bodies ablating in the atmosphere of the earth. The australites then have skipped together with the parent body in order to re-enter as cold spheres. Terrestrial origin of a parent body would require an extremely violent natural event. Ablation analysis shows that fusion of opaque siliceous stone into glass by aerodynamic heating is impossible.

Author

Design Analysis; Tektites; Aerodynamic Heating; Ablation; Aerodynamic Characteristics; Australites

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